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OFFICE OF EXPLORATION

Exploration Studies Technical Report

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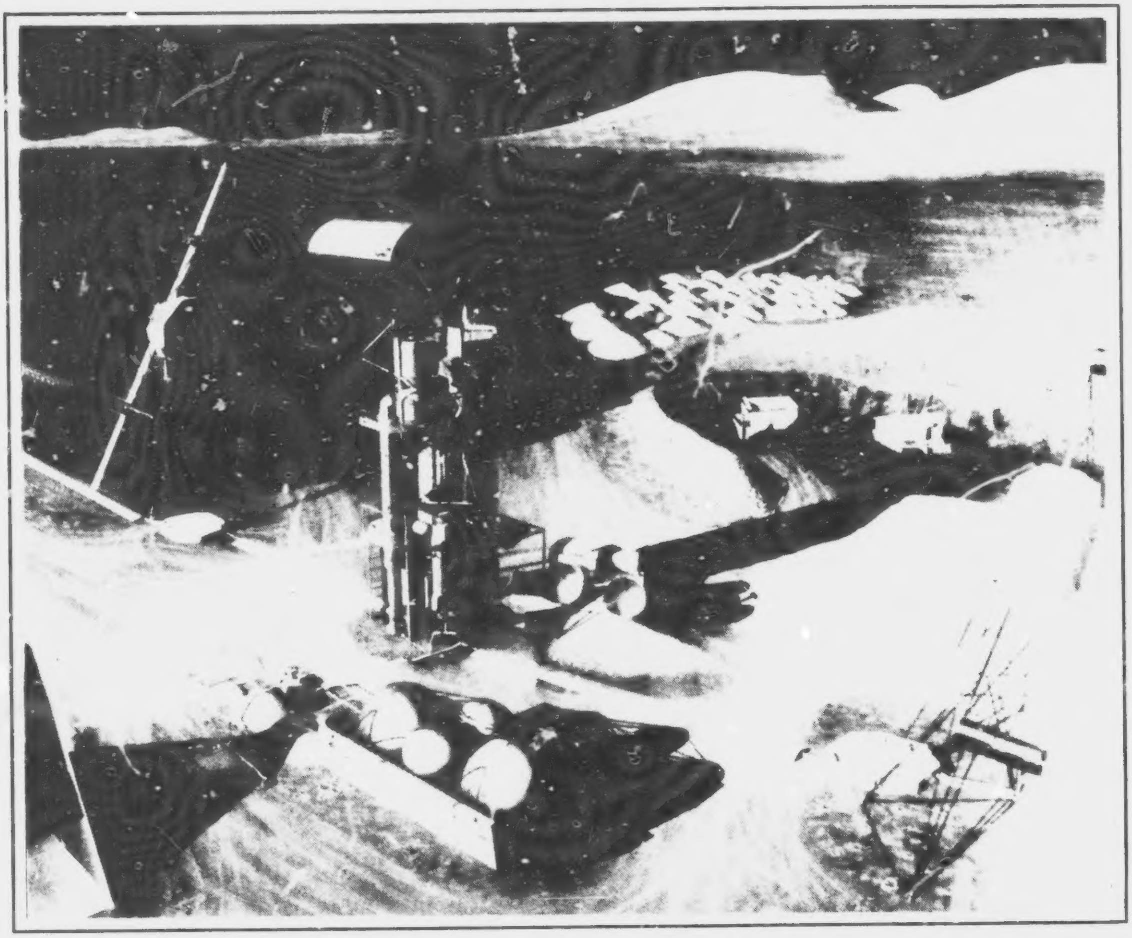

Volume II: Study Approach and Results

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SECTION 1

Exploration Studies Approach

1.1 EXPLORATION STUDY PROCESS OVERVIEW

The NASA Office of Exploration (OEXP) goal is to provide and implement a plan leading to selection of a national space exploration initiative by 1992. This study responds to the OEXP objective of developing options and recommendations for a focused program for human exploration of the solar system. The results of fiscal year (FY) 1988 efforts are described and a framework for more detailed studies in FY 1989 and subsequent years is provided.

Exploration options will require the integrated development and operation of several transportation systems, planetary surface systems, and logistic nodes to achieve the OEXP exploration goal. Some systems already exist such as the National Space Transportation System (NSTS), or are in the process of development such as Space Station Freedom. Others required to support the exploration options will be defined conceptually within this report. The OEXP has developed an annual cyclic process for space exploration study activities. An overview of the process for FY 1988 is shown in figure 1.1-1. The effort involves organizations throughout the Agency as well as academic institutions and industry. OEXP initiates science and user experiment/facility definition studies to identify and define the science potential and objectives for human exploration. The resulting opportunities are sorted by various themes, in order to define exploration strategies which form the basis to initiate the study and synthesis activity. This annual study activity produces technical and programmatic case study descriptions needed to evaluate various exploration alternatives. The case studies define necessary robotic precursors, space

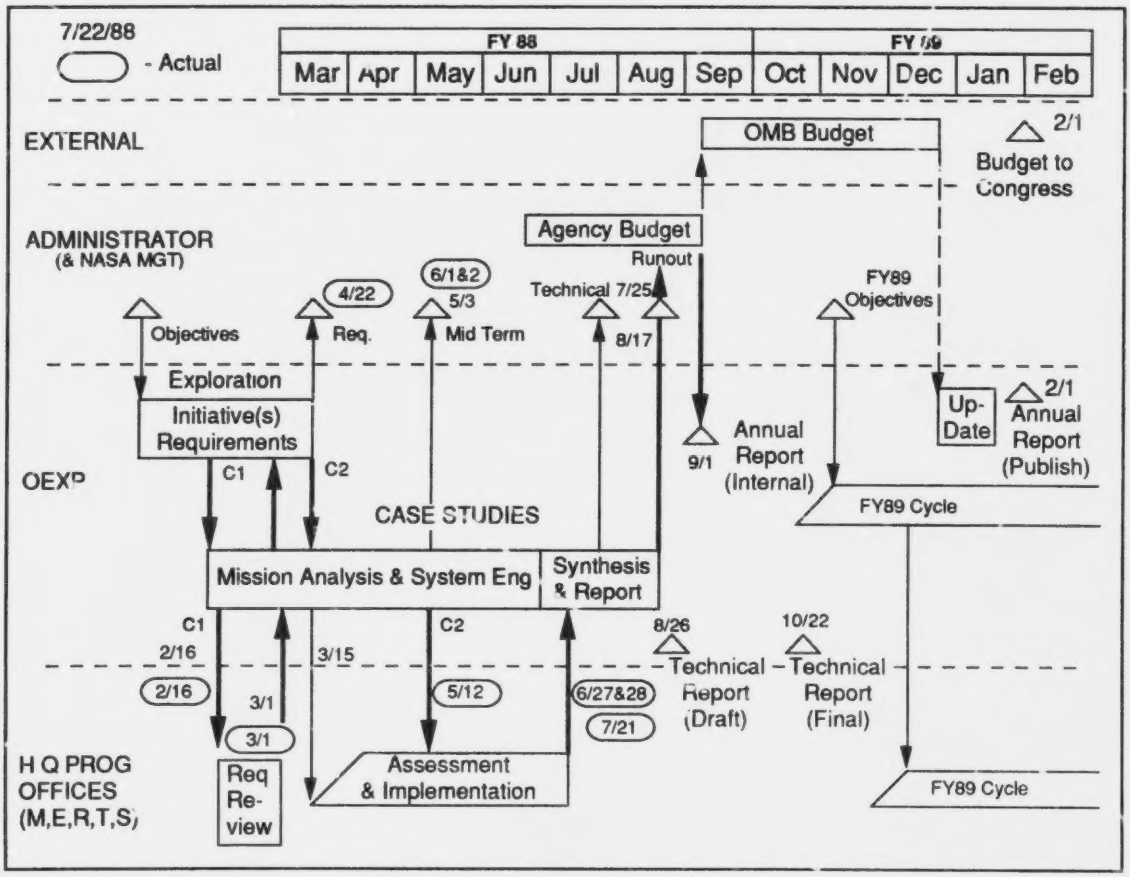
7/22/88
- Actual
FY 88
FY 89
Mar
Apr
May
Jun
Jul
Aug
Sep
Oct
Nov
Dec
Jan
Feb
EXTERNAL
OMB Budget
2/1
Budget to Congress
ADMINISTRATOR (& NASA MGT)
Agency Budget
Runout
Technical 7/25
6/1&2
5/3
4/22
Objectives
Req.
Mid Term
8/17
FY89 Objectives
Exploration
Initiative(s) Requirements
C1
C2
Up-Date
2/1
Annual Report (Publish)
9/1
Annual Report (Internal)
OEXP
CASE STUDIES
FY89 Cycle
Mission Analysis & System Eng
Synthesis & Report
C1
C2
2/16
2/16
3/15
3/1
3/1
5/12
6/27&28
7/21
8/26
10/22
Technical Report (Draft)
Technical Report (Final)
H Q PROG OFFICES (M,E,R,T,S)
Req Re-view
Assessment & Implementation
FY89 Cycle

Figure 1.1-1.- Exploration planning, integration, & reporting process.

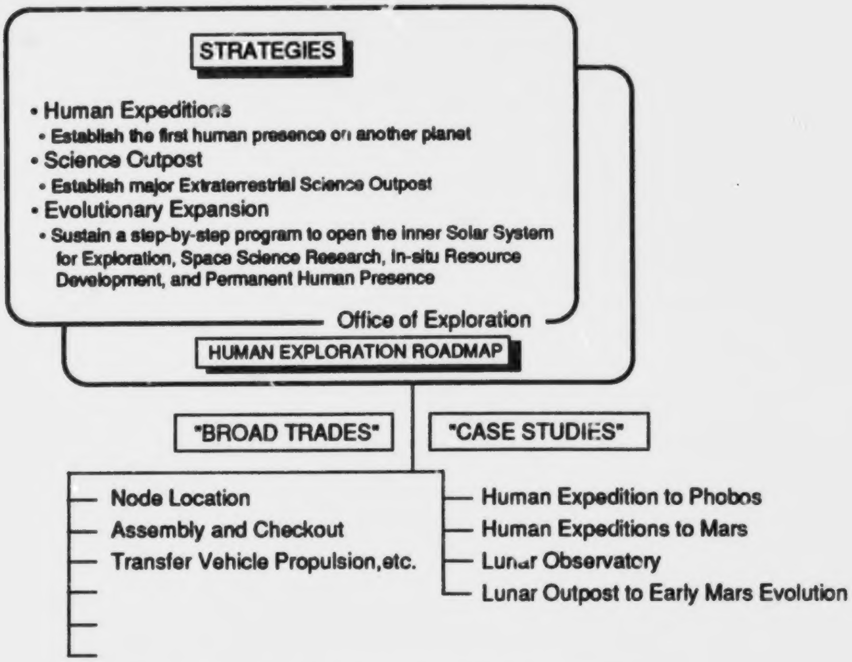
STRATEGIES
• Human Expeditions
• Establish the first human presence on another planet
• Science Outpost
• Establish major Extraterrestrial Science Outpost
• Evolutionary Expansion
• Sustain a step-by-step program to open the inner Solar System for Exploration, Space Science Research, In-situ Resource Development, and Permanent Human Presence
Office of Exploration
HUMAN EXPLORATION ROADMAP
"BROAD TRADES"
"CASE STUDIES"
Node Location
Assembly and Checkout
Transfer Vehicle Propulsion,etc.
Human Expedition to Phobos
Human Expeditions to Mars
Lunar Observatory
Lunar Outpost to Early Mars Evolution

Figure 1.1.1-1.- FY 1988 exploration study approach.

systems, and research and technology needs. These requirements are assessed for implementation by the NASA Headquarters program offices to ensure compatibility with planned science, technology, and operations Research and Development (R&D) budgets.

1.1.1 Exploration Strategies and Options Identification

The process of identifying promising exploration options first requires that top-level strategies be determined (figure 1.1.1-1). These exploration strategies should address the possible human exploration options for the early decades of the twenty-first century and challenge the capabilities of a broad range of technical disciplines by emphasizing the opportunities for space expeditions, exploration, and exploitation.

The primary thrust of the Human Expeditions strategy is to establish the first human presence on another planet. The Science Outpost strategy would establish a major extraterrestrial science outpost. The third strategy, Evolutionary Expansion, would sustain a step-by-step program to open the inner solar system for exploration, space science research, in situ resource development, and permanent human presence.

Figure 1.1.1-2 maps pathways by which candidate exploration options would be used to implement the strategies. As the pathway becomes more complex, the challenge of implementing the necessary technology, engineering, operations, and programmatics increases.

1.1.2 The Case Study Process

The examination of the exploration options is executed through a case study process. The process develops top-level definitions of systems capable of performing the experiments and processes which execute one or more exploration strategies and accommodate the associated requirements within the framework of national space exploration goals. Specifically, the process consists of iterative study phases (figure 1.1.2-1).

The first phase addresses conceptual mission analysis and the associated broad trades. During this effort, mission requirements are defined which meet the exploration goals and objectives (outlined in section 1.2) and user requirements. The mission requirements specify performance parameters for systems defined by this study, identify environments in which the conceptual systems must operate to meet the specified requirements, and point towards the broad trade areas, shown in table 1.1.2-I. The technical options available within each trade area are analyzed for their leveraging value. These trades identify the system concept options and elements which outline the case study.

The second phase of the case study process, systems engineering, performs system-level studies and synthesis using the results of the previous phase. Three domains of interest were identified as significant study areas: transportation systems, orbital nodes, and planetary surface systems, all of which are, in general, programmatically independent and can be addressed ini-

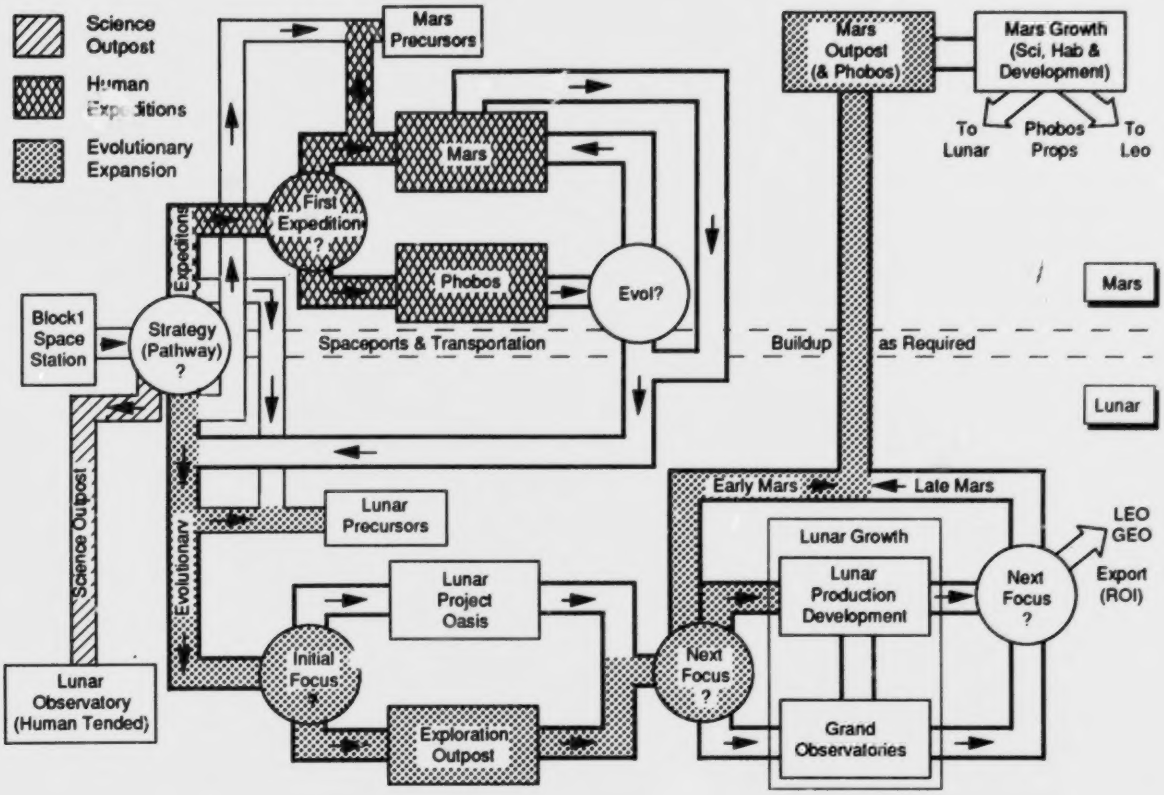
Science Outpost
Human Expeditions
Evolutionary Expansion
Mars Precursors
Mars
First Expedition ?
Phobos
Evol?
Mars Outpost (& Phobos)
Mars Growth (Sci, Hab & Development)
To Lunar
Phobos Props
To Leo
Mars
Block1 Space Station
Strategy (Pathway) ?
Expeditions
Spaceports & Transportation
Buildup as Required
Lunar
Science Outpost
Evolutionary
Lunar Precursors
Early Mars
Late Mars
Lunar Growth
Lunar Project Oasis
Lunar Production Development
Next Focus ?
LEO GEO
Export (ROI)
Initial Focus
Exploration Outpost
Next Focus ?
Grand Observatories
Lunar Observatory (Human Tended)

Figure 1.1.1-2.- Human exploration roadmap.

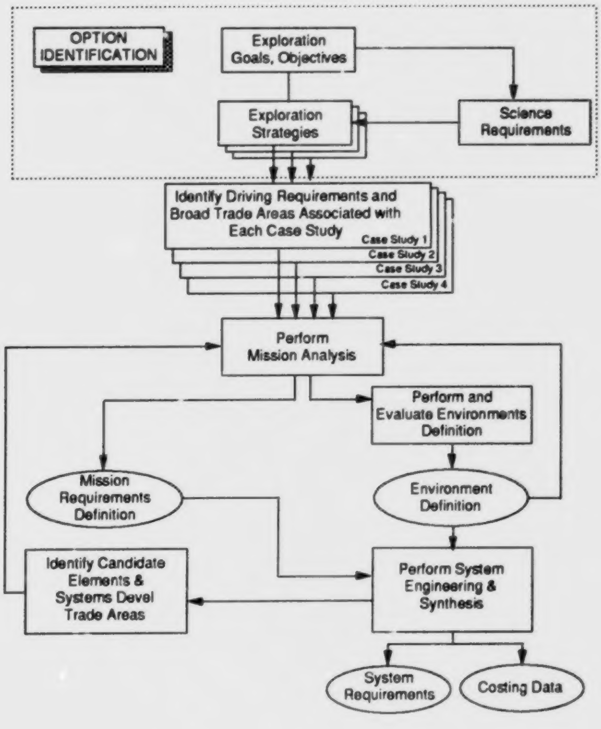
OPTION IDENTIFICATION
Exploration Goals, Objectives
Exploration Strategies
Science Requirements
Identify Driving Requirements and Broad Trade Areas Associated with Each Case Study
Case Study 1
Case Study 2
Case Study 3
Case Study 4
Perform Mission Analysis
Perform and Evaluate Environments Definition
Mission Requirements Definition
Environment Definition
Identify Candidate Elements & Systems Devel Trade Areas
Perform System Engineering & Synthesis
System Requirements
Costing Data

Figure 1.1.2-1.- Case study development process.

tially as functionally independent. A definition of each system was incorporated into the mission-specific case studies.

The third phase comprises a synthesis of the system-level studies, in which systems requirements assumptions provide a basis for the definition of configuration options, and system-level trade studies identify the parametric cost, performance, and risk. The results also establish a preliminary system concept and a reference configuration which is used to refine the study through several iterations. Where unique science and/or technology needs were identified, such as the possible implementation of nuclear spacecraft propulsion, special studies or assessments were made to identify strategies to accommodate those needs. The refined case studies, associated requirements, and leverage value become the knowledge base of exploration path sensitivities (figure 1.1.2-2). The base will be used to define the exploration initiative options, benefits, and risks which lead to the selection and subsequent decision.

The remainder of this section outlines the FY 1988 study objectives, the necessary study support organization, the flowdown of exploration opportunities accommodation into a study requirements hierarchy; and the resulting definition of study products.

TABLE 1.1.2-I.- BROAD TRADE STUDIES

STUDY	MOTIVATION
Node Location	• Choice affects energy, frequency, complexity of transfer operations • High leverage on continuing operations
Assembly and Checkout	• Should investment be made in LEO infrastructure or in ETO infrastructure?
Lunar/PhobosPropellants Leverage	• Using extraterrestrial resources may greatly reduce mass to LEO, but require large investment
Transfer Vehicle Propulsion	• Electric and other advanced technologies could greatly reduce LEO mass requirements
Aerobraked vs. All-Propulsive	• Large effect on LEO mass requirements • Major technology driver
Expendable vs. Reusable Vehicles	• Cost leverage if traffic rate warrants
Zero g vs. Artificial g	• Long-term zero g may not be tolerable

1.2 OBJECTIVES FOR FY 1988

The OEXP has defined the following specific objectives to meet the human exploration goal:

a. To conduct specific human exploration case study development and analysis, concentrating on the science and economic benefit potential for exploration opportunities
b. To identify the prerequisite program requirements related to Earth-to-orbit transportation and space station capabilities needed for each study; and to coordinate the implementation of these requirements with the NASA program offices
c. To identify the prerequisite program requirements related to human space adaptation research
d. To develop the requirements for science and engineering robotic precursor missions
e. To refine the Pathfinder technology requirements necessary to enable and support a range of future exploration strategies
f. To organize and lead the appropriate expertise throughout the Agency and the country in the study and development of potential NASA exploration plans

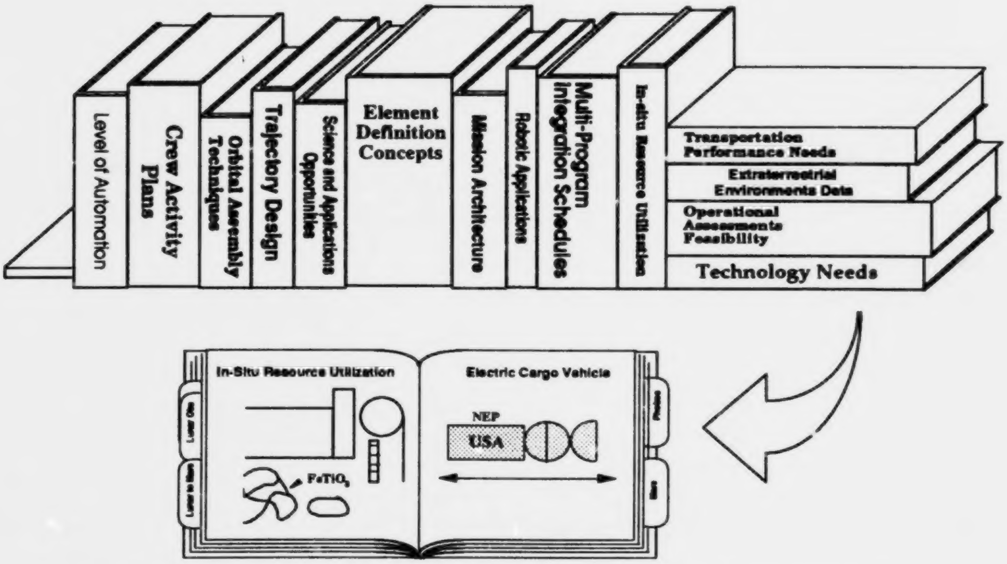
Level of Automation
Crew Activity Plans
Orbital Assembly Techniques
Trajectory Design
Science and Applications Opportunities
Element Definition Concepts
Mission Architecture
Robotic Applications
Multi-Program Integration Schedules
In-situ Resource Utilization
Transportation Performance Needs
Extraterrestrial Environments Data
Operational Assessments Feasibility
Technology Needs
In-Situ Resource Utilization
$FeTiO_3$
Electric Cargo Vehicle
NEP
USA

Figure 1.1.2-2.- Building a knowledge base of exploration pathway sensitivities.

This study report describes the initial effort of implementing the objectives during FY 1988 and the resulting plans for FY 1989. Within the scope of the FY 1988 study activity, the following questions were addressed:

a. How do the diverse exploration stategies embodied in the case studies compare when developed with consistent methodology and to a uniform level of detail?
b. What are the key parameters affecting each case study with respect to scale, complexity, and achievability?
c. What advantages can be realized by utilizing advanced technologies and/or extraterrestrial resources?.
d. Is it possible to undertake a major space expedition without the use of a low-Earth-orbit (LEO) assembly node or advanced technology?
e. Are the case study needs compatible with existing NASA program plans and schedules?

Study observations related to each of these questions are contained in Section 6.0 of this report.

1.3 OEXP SUPPORT FUNCTIONS DISTRIBUTION

The OEXP established and manages the organization which executes exploration studies. This organization is composed of elements of the existing NASA Headquarters program offices and dedicated NASA center exploration offices which provide the key study functions described in section 1.1. The study functions and associated center assignments are described in the sections that follow.

1.3.1 Science Requirements Synthesis

OEXP conducts survey workshops with elements of the science and user community to understand the key objectives or opportunities for each exploration concept. The function validates the science and user opportunities through integrated analyses of the objectives by a broad based science and user community. A discussion of the treatment of this function for the FY 1988 study is contained in section 1.4.

1.3.2 Mission Analysis and Systems Engineering

Johnson Space Center (JSC) performs all case study development and analysis, including the coordination and synthesis of all domain studies, technology, and special assessments. The function includes documentation of the technical and programmatic requirements resulting from the case studies.

1.3.3 Domain-Level Studies

Integration Agents (IA's) perform systems concept studies and trades, and develop configurations, requirements, and implementation plans for the following domains:

a. Space Transportation Systems—Marshall Space Flight Center (MSFC)
b. Nodes—Langley Research Center (LaRC)
c. Planetary Surface Systems—JSC

1.3.4 Special Assessments and Studies

Special Assessment Agents (SAA's) perform discipline-related studies relevant to technological, operational, or programmatic needs as defined in the case study development process. The studies are related to IA activities in that a specific high-leverage discipline area is identified in a study domain. The SAA will focus the discipline area to provide leverage, risk, and potential implementation details. The following specific assessment areas have been addressed for FY 1988:

a. Power Systems—Lewis Research Center (LeRC)
b. Propulsion Systems—LeRC
c. Automation & Robotics/Expert Systems—Ames Research Center (ARC)
d. In-Space Assembly Operations—JSC
e. Advanced Life Support Systems—JSC
f. Exploration Cost Understanding & Methodology—JSC

1.4 ACCOMMODATION OF EXPLORATION OPPORTUNITIES

Table 1.4-I illustrates a set of generalized science and exploration objectives. For purposes of evaluation, this set is the basis for development of case-specific science payloads. For the FY 1988 study, the science objectives analysis outlined in section 1.3.1 had not yet been initiated. A synthetic set of science payloads was created to size the study requirements and complete the first cycle process. As the science objectives mature, their resulting payloads will be incorporated into the refined case study results in FY 1989. A further discussion of exploration opportunities and benefits is found in section 4.10.

The accommodation of the exploration opportunities for each case study is outlined in sections 2.1, 2.2, 2.3, and 2.4.

1.5 STUDY REQUIREMENTS HIERARCHY

The OEXP has identified and produced documentation which controls the exploration study process, associated requirements, and results described in section 1.1. The hierarchical relationship of the study requirements documentation is shown in figure 1.5-1. The OEXP has produced and controlled the Exploration Requirements Document (ERD), which controls the case study development activity by providing top-level requirements as well as ground rules and assumptions to guide the study process.

TABLE 1.4-I.- GENERALIZED SCIENCE AND EXPLORATION OBJECTIVES

- To study the origin, history, and current state of planetary bodies of the solar system and to understand their relation to the Earth and the origin of the solar system
- To seek evidence for the origin and evolution of living organisms through identification of environments in which life could have existed or through identification of physical or chemical remains
- To conduct studies of the universe that can be uniquely or effectively undertaken utilizing the new environments that would be accessible in the human exploration program
- To utilize newly significant accessible environments to conduct studies of importance to other fields of science (e.g., high vacuum on the Moon or at a libration point; 1/6 or 1/3 g)
- To understand the abilities and limitations of human beings for extended tours of duty in the new environments, particularly long-duration space flight beyond Earth orbit and on planetary surfaces
- To establish the feasibility and utility of permanent human outposts on the surfaces of other planets

The ERD precipitates the Study Requirements Document (SRD), which contains detailed technical information regarding the individual case studies and facilitates the necessary IA and SAA studies. The initial baseline of the SRD contains hypothetical data and requirements developed by the JSC Mission Analysis and System Engineering (MASE) group in its preliminary system engineering activities. As the IA's technical analyses improve the pedigree of data available to the MASE process, the SRD will be updated to drive the next study cycle.

The OEXP also produces the Prerequisites Requirements Document (PRD). Its purpose is to provide the mechanism by which the exploration-enabling system, environmental knowledge, and life science and technology requirements are transmitted to the various NASA program offices. The program offices, in turn, define and document specific approaches to implement the requirements.

1.6 STUDY RESULTS

The subsequent sections of this report describe the work accomplished within the initial analysis cycle and discuss opportunities for additional study. Section 2.0 discusses the technical results of each of the four case studies. Included in the discussion of each case study is a summary description of the mission key features and profile. Mission definition and manifesting are detailed, followed by a description of the mission architecture and

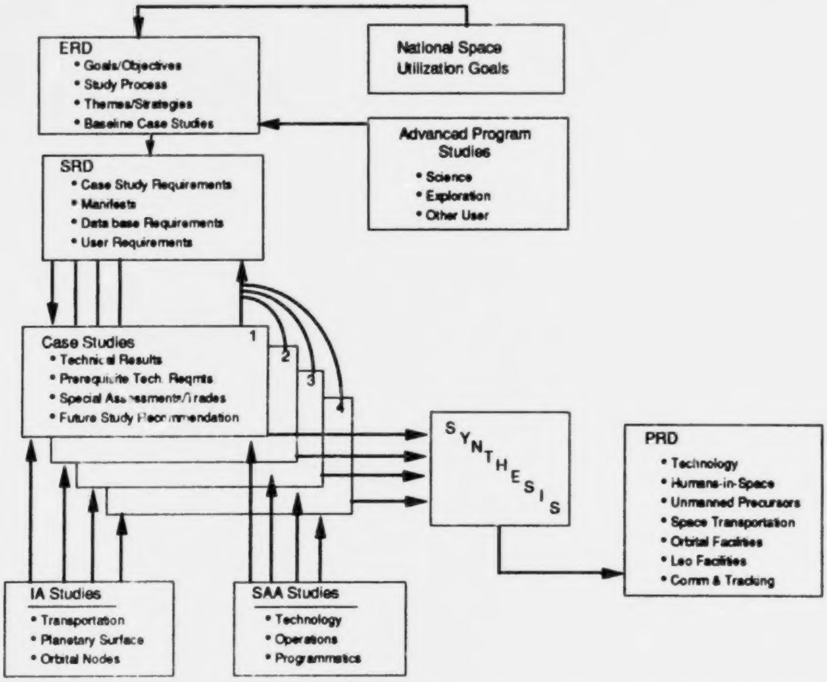
ERD
• Goals/Objectives
• Study Process
• Themes/Strategies
• Baseline Case Studies
National Space Utilization Goals
Advanced Program Studies
• Science
• Exploration
• Other User
SRD
• Case Study Requirements
• Manifests
• Data base Requirements
• User Requirements
Case Studies
• Technical Results
• Prerequisite Tech. Reqmts
• Special Assessments/Trades
• Future Study Recommendation
1
2
3
4
SYNTHESIS
PRD
• Technology
• Humans-in-Space
• Unmanned Precursors
• Space Transportation
• Orbital Facilities
• Leo Facilities
• Comm & Tracking
IA Studies
• Transportation
• Planetary Surface
• Orbital Nodes
SAA Studies
• Technology
• Operations
• Programmatics

Figure 1.5-1.- Relationship of exploration study requirements documentation.

infrastructure. Systems concept discussions follow for the required orbital nodes, transportation systems, and planetary surface systems. A synthesis of each case study evaluates the results against defined criteria. The synthesis is summarized by the identification of principal issues and program risks.

Section 3.0 deals with the prerequisite implementation plans resulting from the synthesized case studies. After reviews of these study results, the major NASA program offices identify the exploration impacts to their plans. Case study program integration schedules are then defined.

Special reports, trade study results, and indepth system assessments are reported in sections 4.0 and 5.0. The scope of the special reports and trade study activity includes Earth-Moon node location, extraterrestrial propellant leveraging, LEO assembly strategy, lunar mining of fusion fuels (He-3), transfer vehicle propulsion options (e.g., electric propulsion for cargo vehicles), extended crew lunar staytime, robotics site preparation, and space exploration cost explication.

Indepth system assessments were performed in areas involving power, propulsion, and life support systems. A major result has been the value of nuclear thermal rocket (NTR) propulsion and nuclear lunar base power systems. The reduced propellant and trip time advantages of solid-core NTR in Nuclear Engine for Rocket Vehicle Application (Nerva) class and gas-core NTR's (both open- and closed-cycle) are discussed, as is the maturity level of each technology. A similar treatment of an SP-100 class power system is provided for satisfying lunar base power requirements.

The studies, trades, and system assessments in sections 4.0 and 5.0 are largely case study independent and were not subject to the input requirements and constraints levied upon the case studies analyses agents. Therefore, conclusions reached may vary from or even contradict those related to specific case studies.

Section 6.0 discusses overall technical and programmatic observation case study activities. These observations are arranged in categories of lessons learned, case studies comparative analysis, and key findings related to current and planned programs including prerequisite program needs.

SECTION 2

Analysis and Conclusions of Case Study Results

This section summarizes the results of work on the four case studies selected for investigation during FY 1988:

Human Expedition to Phobos
Human Expeditions to Mars
Lunar Observatory
Lunar Outpost to Early Mars Evolution

The four case studies are described in the following sections, 2.1 through 2.4.

Sections 2.X.1 through 2.X.3 for each case study were prepared by the Mission Analysis and System Engineering (MASE) agent. They describe the major features of the case study, including the mission profile (sequence of events, timeline, etc.) and pertinent summary data such as mass and crew size. The material in these sections reflects the baseline case study as defined in the Exploration Requirements Document (ERD) and the Study Requirements Document (SRD).

Sections 2.X.4, 2.X.5, and 2.X.6 were prepared by the Transportation, Orbital Node, and Planetary Surface Systems Integration Agents (IA's) respectively. These sections detail the element concepts and configurations that fall within each IA's domain. Each section includes a description of the baseline system, any required technology advances that were identified within the IA's area, and any alternative approaches to the case study that were developed during the investigation.

The last section, 2.X.7, in each case study is a MASE synthesis of the inputs from the IA's and Special Assessment Agents (SAA's). Emphasis is on evaluation of technical continuity of the end-to-end case study when the piece-part IA and SAA study inputs are considered as an integrated set. Unresolved issues and program risks are identified. Evaluation of the data provided by the IA's and SAA's was a necessary first step in the synthesis. In selecting the criteria by which this evaluation would be carried out, two general guidelines were applied: (1) whether the data fit within the case study as a whole, and (2) whether an objective evaluation of the data could be made.

The specific criteria selected were:

a. Consistency - The degree of technical consistency in study assumptions and results between Transportation, Node, and Planetary Surface Systems IA's.

b. Parametric Results - The degree to which IA's and SAA's parameterized/scaled their study results to facilitate adjustments to the scope and mission objectives of the case study.

c. Options - The degree to which options for implementing the baseline or case study alternatives were identified and analyzed.

d. Special Assessments and Broad Trades - The degree to which Broad Trade and Special Assessment studies were relevant to the case study results, and the degree to which the IA's were able to use those results to achieve case study objectives more effectively.

In addition to direct support for synthesis activities, the SAA's also provided independent analyses as deemed appropriate. An excellent example for FY 1988 is the indepth assessment study dealing with high leverage gas-core nuclear thermal rocket technology provided by the Propulsion SAA (Lewis Research Center) and included as section 5.2.3 of this report.

By its nature, the evolutionary Case Study 4 is highly complex in its long-term mission objectives, its requirements for development and use of advanced technology, and its requirement for a permanent lunar and Mars space systems infrastructure to sustain the crew and mass flow rate crucial to the success of this case study. As a result, the ERD and SRD requirements matured too late in FY 1988 to receive an indepth analysis by the IA's and SAA's. Consequently, the mission design, element configuration definition, and overall case study synthesis activity reported in section 2.4.7 were accomplished within the MASE function. Detailed analysis by the IA's and SAA's is planned for FY 1989.

2.1 HUMAN EXPEDITION TO PHOBOS (CASE STUDY 1)

In this case study the prime mission objective is the establishment of early leadership in manned exploration of the solar system. To that end the baseline vehicles are designed for minimum dependence on advanced technology, and human presence is extended only to Mars orbit and the surface of Phobos, rather than the surface of the planet itself.

2.1.1 Case Study Overview

2.1.1.1 Key Features

Along with the orbital departure maneuvers at Mars and Earth, capture into Mars orbit is accomplished by state-of-the-art chemical rockets. At the end of their round trip, the four-person flightcrew is returned directly to the Earth surface by an Apollo-type atmospheric entry module.

The surface of Mars is explored by rovers teleoperated by the flightcrew from Mars orbit, and Phobos is explored by human extravehicular activity (EVA). All interplanetary vehicles are designed to be expendable, and no provisions are made for artificial gravity. Nominally, the flightcrew arrives in (and departs from) the vicinity of Mars during the year 2003. The next opportunity, which was considered as a backup, occurs in 2005.

2.1.1.2 Mission Profile

The heliocentric trajectory profile is referred to as "split/sprint." This signifies that the interplanetary payload is split between a cargo carrier and a separate crew carrier, and that the heliocentric trajectory profile of the crew carrier belongs to a subset of opposition-class round-trip flightpaths which have been characterized as sprint trajectories.

For the crew carrier the stay time in Mars orbit was chosen to be 30 days, and the round-trip time was limited to 440 days (which is shorter than optimum for the average mission opportunity), with a view toward reducing the severity of problems associated with human factors and system reliability. A maximum atmospheric entry speed of 12.2 km/s was imposed as a constraint during the selection of return trajectories for the crew carrier. This entry speed is consistent with known (Apollo) technology.

The cargo carrier is launched from low Earth orbit (LEO) on a minimum energy (near-Hohmann) heliocentric transfer trajectory about 18 months before Earth departure time for the crew carrier. Requiring much less delta V than the crew carrier, it carries all supplies and equipment not vital to the crew on their outbound journey to the vicinity of Mars. The most important part of the cargo carrier payload is the propellant needed by the crew carrier for departure from Mars orbit on the inbound leg of the sprint trajectory. Other cargo items include teleoperated Mars surface rovers, Mars orbital science and communication satellites, and an excursion vehicle designed to transport two crewmembers to the surface of Phobos.

Upon arrival at Mars, several months before the flightcrew leaves Earth, the cargo carrier is injected into a 250 by 33,840 km areocentric ellipse which later is circularized at the higher altitude. Using propulsion to adjust orbit altitude and thus control the rate of nodal regression due to Mars oblateness, the plane of the cargo carrier's orbit is established in such an orientation that it will contain both the arrival and the departure hyperbolae of the crew carrier, thus avoiding plane-change penalties against that vehicle.

The inclination of the parking orbit relative to the Martian equator varies with mission opportunity, being about 27° in 2003 and 34° in 2005. The status of the cargo carrier is monitored by radio while awaiting rendezvous with the crew carrier, and the satellites and Mars surface rovers carried by it from Earth are deployed at appropriate times before arrival of the flightcrew.

The crew carrier departs LEO about 7 months before the minimum-energy departure opportunity that succeeds the one used by the cargo carrier. Its outbound flightpath traverses a heliocentric angle of approximately 300° in about 8 or 9 months and takes it briefly inside the orbit of Venus, thus acquiring the phase angle necessary for an early return to Earth. For the nominal (2003) Mars mission opportunity, Venus is in a favorable position to be used for gravitational shaping of the outbound heliocentric trajectory, and is so employed. Venus is not well situated for such a purpose in the backup (2005) opportunity, but a similar kind of heliocentric trajectory shaping is achieved by a deep-space propulsive manueuver.

Upon arrival at Mars the crew carrier is first injected into a 250 by 33,840 km orbit, which is later circularized at 33,840 km during rendezvous with the cargo carrier. Following crew/cargo rendezvous, the crew carrier docks with the trans-Earth injection (TEI) stage that was transported to Mars by the cargo vehicle, and a few days later two of the crewmembers enter the Phobos excursion vehicle. The excursion vehicle leaves the main spacecraft and carries them through plane-change, altitude adjustment, and phasing maneuvers to a rendezvous with Phobos in its near-circular, near-equatorial orbit 5980 km above the surface of Mars.

The Phobos exploration crew spends 20 days in the near vicinity of that martian satellite, and in that time they make observations, conduct experiments, and gather samples during a total of 24 hours of EVA. Besides monitoring the Phobos exploration activities, the two other crewmembers in the main spacecraft also teleoperate the two rovers which gather samples of material from the surface of Mars.

Several days before Mars departure time, the Phobos exploration crew flies the excursion vehicle to a rendezvous with the main spacecraft. The excursion vehicle is jettisoned after crew and samples have been transferred to the main spacecraft. At appropriate times the ascent stages carried by the unmanned rovers also deliver their surface-sample payloads to the main spacecraft, after which they also are jettisoned.

Shortly before the scheduled departure time, and at a position opposite the optimum location of periapse on the Mars departure hyperbola, a retrograde velocity increment of about 620 m/s is applied to the interplanetary crew carrier vehicle. This sets up an Oberth departure maneuver by establishing an elliptical periapse at an altitude of 250 km, where the final TEI impulse is applied.

Returning home on a trajectory segment that is bounded by the orbits of Mars and Earth, the crew carrier traverses a heliocentric angle of approximately 115° in 4 or 5 months. Some hours before the main spacecraft passes by Earth at an altitude of a few hundred kilometers, the flightcrew joins the previously-stowed samples of Mars and Phobos surface material, in an Apollo- type atmospheric entry module. After being separated from the main spacecraft, the entry module is nudged by a small propulsive velocity increment which alters its trajectory so that it will enter the atmosphere at a flightpath angle appropriate for its speed and land in a preselected recovery area.

2.1.1.3 Summary Data

Table 2.1.1-I contains a summary of major trajectory parameters for the two mission opportunities that were studied. Table 2.1.1-II contains corresponding LEO departure masses for the interplanetary vehicles. In the second table, mass data are shown for three of the more attractive alternatives to the baseline design. These are discussed in more detail in section 2.1.4.3.

The planetary departure and arrival delta V values shown in table 2.1.1-I represent impulsive velocity increments for departure from and capture into optimally oriented planetocentric orbits. The predeparture orbit altitude at Earth was assumed to be 500 by 500 km. The arrival and departure increments at Mars represent the periapsidal components of Oberth capture and escape maneuvers. They do not include the 619 m/s required to circularize after the initial capture into a 250 by 33,840 km ellipse, nor the like amount required to lower the periapse altitude just before the final TEI impulse.

The data in table 2.1.1-I do not include allowances for an orbital departure window at either planet, nor for gravity losses and performance reserves. Nominal performance reserves were accounted for in the computation of the masses shown in table 2.1.1-II but, again, those data do not reflect allowances for launch windows nor for gravity losses.

TABLE 2.1.1-I.- TRAJECTORY DATA FOR PHOBOS EXPEDITION

Vehicle	Event/Parameter	Mission Opportunity	
		2003*	2005*
Cargo Carrier	**LEO Departure**		
	Elapsed Days	0	0
	Date	2001 Feb 04	2003 Jun 07
	Decl Vinf (deg)	-6	-6
	Delta V (m/s)	3737	3555
	Mars Arrival		
	Elapsed Days	268	202
	Date	2001 Oct 30	2003 Dec 26
	Decl Vinf (deg)	-29	7
	Delta V (m/s)	1588	924
Crew Carrier	**LEO Departure**		
	Elapsed Days	0	0
	Date	2002 Aug 15	2004 Nov 19
	Decl Vinf (deg)	-28	-15
	Delta V (m/s)	4350	4896
	Venus Swingby		
	Elapsed Days	80	
	Date	2002 Nov 03	
	Solar Dist (au)	0 72	
	Min Alt (km)	3412	
	Deep-Space Mnvr		
	Elapsed Days		82
	Date		2005 Feb 09
	Solar Dist (au)		0 70
	Delta V (m/s)		1614
	Mars Arrival		
	Elapsed Days	286	254
	Date	2003 May 28	2005 Jul 30
	Decl Vinf (deg)	-24	4
	Delta V (m/s)	3936	3164
	Mars Departure		
	Elapsed Days	316	284
	Date	2003 Jun 27	2005 Aug 29
	Decl Vinf (deg)	-20	28
	Delta V (m/s)	2335	3259
	Earth Return		
	Elapsed Days	440	440
	Date	2003 Oct 29	2006 Feb 02
	Decl Vinf (deg)	22	1
	Entry V (m/s)	11756	11482

*Crew arrival dates at Phobos

TABLE 2.1.1-II.- LEO DEPARTURE MASS FOR PHOBOS EXPEDITION

Design Option	Mission Opportunity		LEO Departure Mass Component
	2003*	2005*	
Baseline	468 t	317 t	Cargo Carrier
	1311 t	2048 t	Crew Carrier
	1779 t	2365 t	Total
Nuclear Thermal Rockets NTR	314 t	—	Cargo Carrier
	538 t		Crew Carrier
	852 t		Total
Mars Aerocapture	312 t	275 t	Cargo Carrier
	453 t	817 t	Crew Carrier
	765 t	1092 t	Total
NTR + Mars Aerocapture	215 t	—	Cargo Carrier
	282 t		Crew Carrier
	497 t		Total

* Crew arrival dates at Phobos

The software used to select the heliocentric trajectories in this study minimizes a figure of merit formed by summing propulsive velocity increments at some points with V-infinity magnitudes at other points on the flightpath, rather than (better) the sum of all propulsive velocity increments or (better yet, but much more tedious) the LEO departure mass of the spacecraft. The calculated spacecraft masses probably could be reduced to some extent by selecting different heliocentric trajectories but, pending the availability of more suitable software, the magnitude of any such reduction is uncertain. At least it can be said that any effects of nonoptimal trajectory selection will tend to compensate for the omission of launch window allowances.

2.1.2 Mission Definition and Manifest

Flight operations supporting the Phobos expedition include zero-g countermeasures research assumed to begin in 1997. Buildup of the cargo carrier vehicle in LEO begins in the year 2000, preparatory to its departure from LEO early in 2001. Buildup of the crew carrier vehicle begins sometime later in 2001, in preparation for its departure in 2002.

Table 2.1.2-I contains a list of payload items and the quantity of each item to be delivered to its appropriate destination, during each year of the 7-year period of flight operations assumed for this case study.

Mars Surface Systems Element Name	Mass kg	LEO Departure Dates					
		98	99	00	01	02	03
Teleoperated Mars Rovers	3500				2		

Phobos Surface Systems Element Name	Mass kg						
		98	99	00	01	02	03
Space Suits	136.00				2		
MMU's	153.00				2		
EVA Retriever	480.00				1		
Exploration Tools	100.00				1		
Mobility Aids/Restraints	100.00				1		

Mars Orbit Systems Element Name	Mass kg	LEO Departure Dates					
		98	99	00	01	02	03
Communication Satellite(s)	1000.00				1		
Science Satellite(s)	1000.00				1		
Mars Orbital Science Equipment	150.00				1		
Cargo Veh Interplanetary Science Eqpt.	300.00				1		
Pilot Veh Interplanetary Science Eqpt.	500.00						1

Transportation Systems Element Name	Mass t	LEO Departure Dates					
		98	99	00	01	02	03
Cargo Mission: TMI Stage(s) (dry)	40.50				1		
TMI Cryo. Props. (includes MOC, MOO)	326.40				1		
Storable Props. (includes MCC, RCS)	6.80				1		
Cargo Mother Ship	13.60				1		
TEI stage (dry)	8.88				1		
TEI Cryo. Props.	51.90				1		
Crew Mission: TMI Stage(s) (dry)	123.00						1
TMI Cryo. Props. (includes MOC, MOO1)	1046.00						1
Storable Props. (includes MCC, RCS)	15.80						1
Crew Mother Ship	104.50						1
TEI Preparation Cryo. Props. (MOO2)	21.50						1
Phobos Excursion Vehicle (dry minus payload)	2.69						1
Phobos Excursion Vehicle Propellant	6.04						1

TABLE 2.1.2-I.- HUMAN EXPEDITION TO PHOBOS
PAYLOAD ELEMENT MANIFEST

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2.1.3 Mission Architecture and Infrastructure

The mission architecture for this case study is illustrated in figures 2.1.3-1 and 2.1.3-2. The STS Shuttle is used for Earth-to-orbit (ETO) transportation of flight crews, and heavy-lift launch vehicles (HLLV's) are used for all other ETO transportation.

Transportation requirements for the Phobos expedition program are shown graphically in figure 2.1.3-3. Maximum ETO mass in any one year is about 1300 metric tons in 2002, for the most part consisting of cryogenic propellant for the interplanetary crew carrier, which arrives at Mars in 2003 and returns to Earth 440 days after departure from LEO.

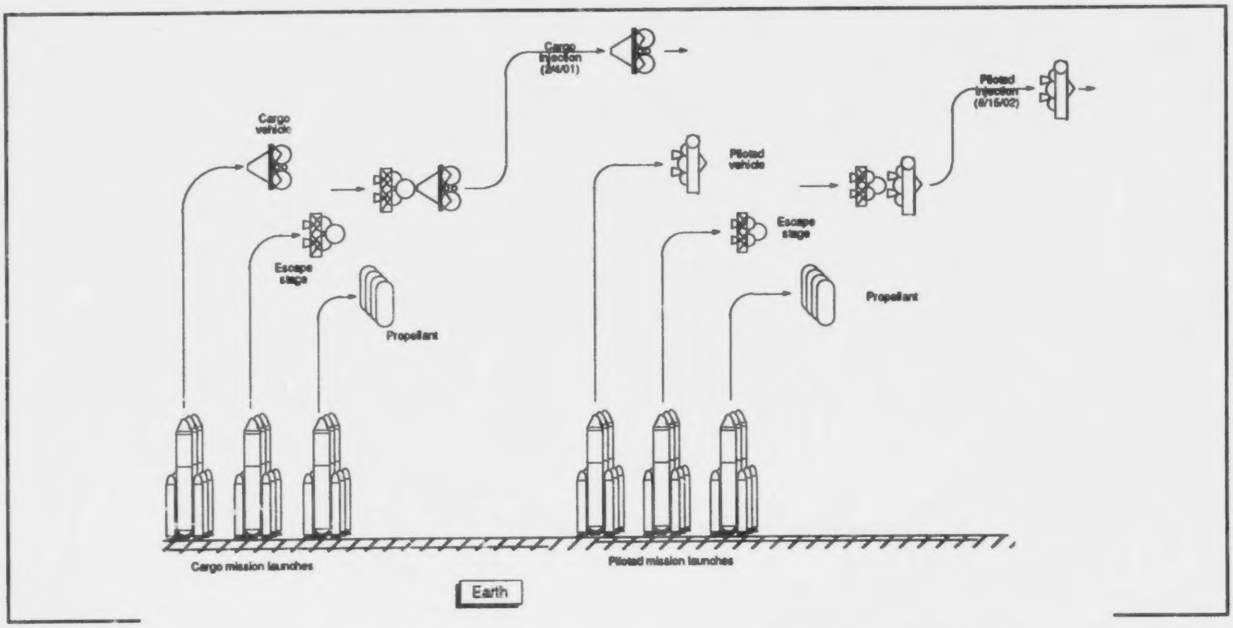

Cargo injection (2/4/01)
Piloted injection (8/15/02)
Cargo vehicle
Piloted vehicle
Escape stage
Escape stage
Propellant
Propellant
Cargo mission launches
Piloted mission launches
Earth

Figure 2.1.3-1.- Human expedition to Phobos — Earth orbital operations.

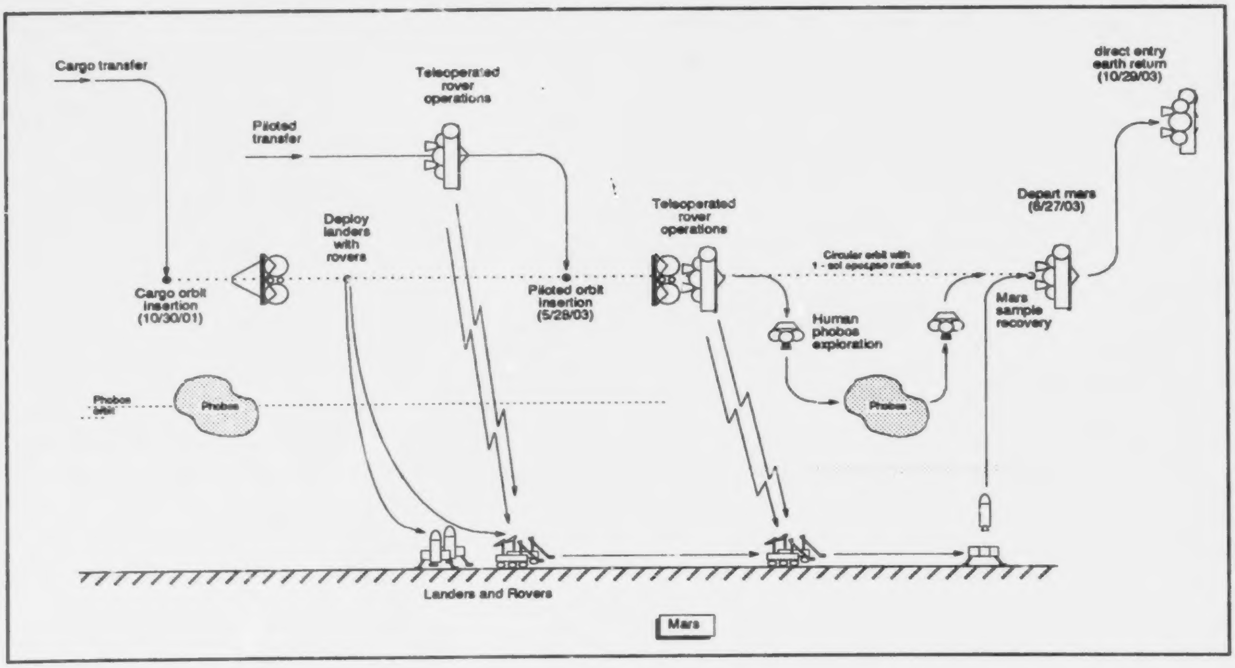
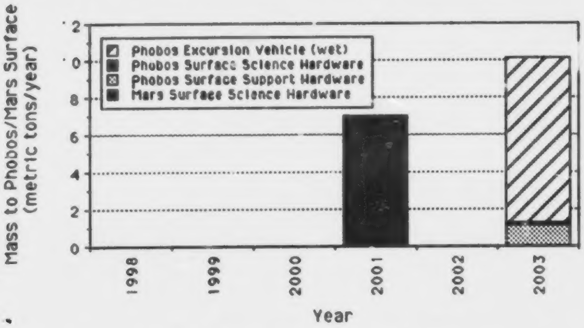

Cargo transfer
Teleoperated rover operations
direct entry earth return (10/29/03)
Piloted transfer
Depart mars (6/27/03)
Deploy landers with rovers
Teleoperated rover operations
Circular orbit with 1 - sol apoapse radius
Cargo orbit insertion (10/30/01)
Piloted orbit insertion (5/28/03)
Human phobos exploration
Mars sample recovery
Phobos orbit
Phobos
Phobos
Landers and Rovers
Mars

Figure 2.1.3-2.- Human expedition to Phobos — Mars/Phobos orbital operations.

Human Expeditions to Phobos



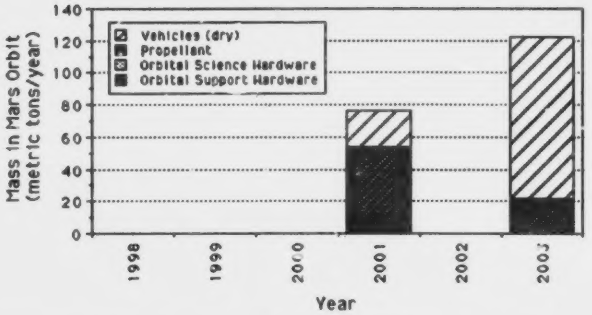
Note: Mass to Phobos/Mars Surface

Phobos Excursion Vehicle (wet) includes the wet Phobos excursion vehicle minus any payload

Phobos Surface Science Hardware includes the equipment used to conduct the science experiments on Phobos e.g., exploration tools)

Phobos Surface Support Hardware includes the equipment used to support the science experiments on Phobos (e.g., space suits, MMU's, mobility aids and restraint)

Mars Surface Science Hardware includes the teleoperated Mars rovers



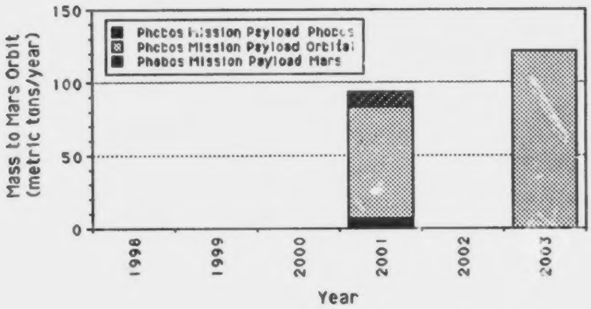
Note: Mass in Mars Orbit (this includes the things that are used in Mars orbit)

Vehicles (dry) includes the dry trans-Earth injection stage, the dry crew mother ship, and the dry cargo mother ship

Propellant includes the cryogenic propellant used to return to Earth

Orbital Science Hardware includes the equipment used to conduct the science experiments in Mars orbit (e.g., Mars science satellites)

Orbital Support Hardware includes the equipment used to support the science experiments in Mars orbit (e.g.., Mars communications satellites)

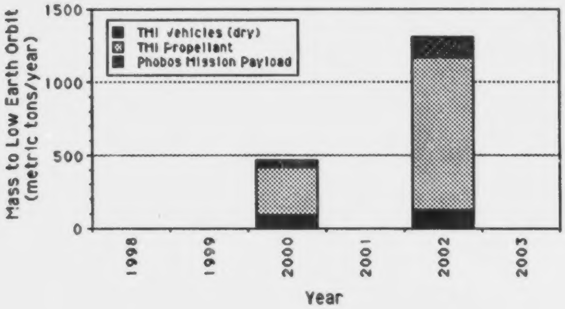


Note: Mass to Mars Orbit (this includes everything taken to Mars)

Phobos Mission Payload - Phobos includes the wet Phobos excursion vehicle and all science and support hardware taken to Phobos

Phobos Mission Payload - Orbit includes the wet trans-Earth injection stage, the crew and cargo mother ships, and all science and support hardware used or deployed in orbit

Phobos Mission Payload - Mars includes the teleoperated Mars rovers



Note: Mass to Low Earth Orbit

TMI Vehicles (dry) includes the dry trans-Mars injection stages and the storable propellants and crew consumables used in getting to Mars

TMI Propellant includes the cryogenic propellant used to get the cargo and crew vehicles to Mars

Phobos Mission Payload includes the crew and cargo motherships, the cryogenic propellant used to return the crew to Earth, the Phobos excursion vehicle, and all science and support hardware used at Phobos, in Mars orbit, and on the martian surface

Figure 2.1.3-3.- Human expedition to Phobos — Transportation requirements

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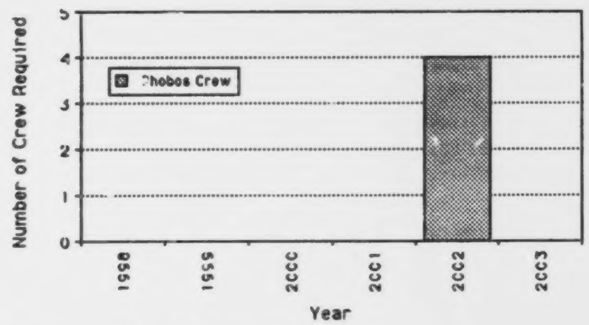

Note: Number of Crew Required

Phobos Crew is the four-person crew sent to Phobos

Figure 2.1.3-3.- Concluded.

The 440-day sprint mission opportunity for Phobos exploration in 2003 is very unusual in that (1) the crew carrier arrives when Mars is at the optimum location in its elliptical heliocentric orbit for such missions, and (2) Venus is favorably located for gravitational shaping of the crew carrier's heliocentric trajectory by means of a Venus swingby (VSB). Each of these two circumstances by itself has a very significant effect in reducing the propulsion requirements for a sprint mission. Combined as they are in the 2003 opportunity, they make for a very favorable situation that occurs only once in every 15 mission opportunities (32 years).

Major milestones for the human expedition to Phobos are shown in figures 2.1.3-4 and 2.1.3-5.

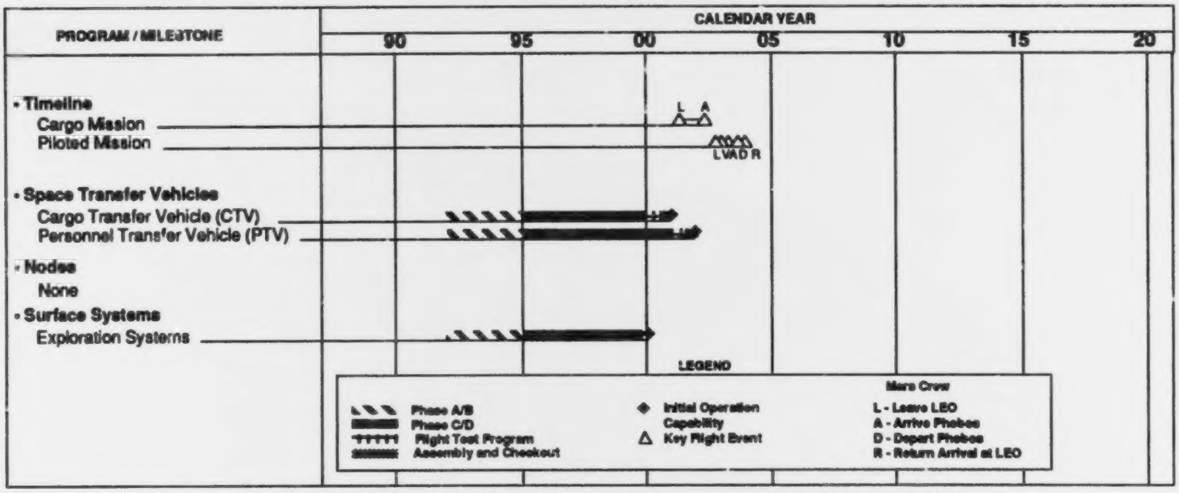

Figure 2.1.3-4.- Milestones for human expedition to Phobos.

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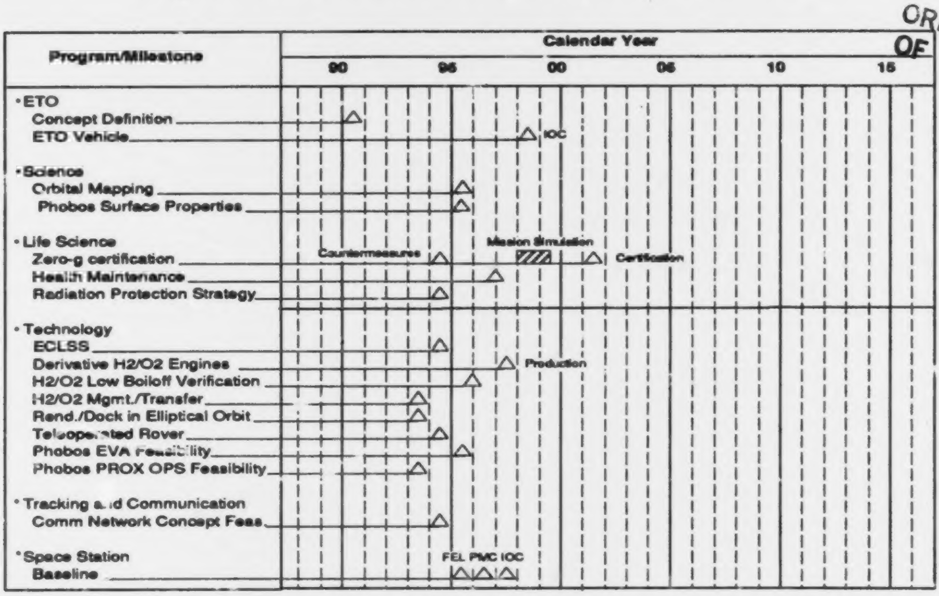


Figure 2.1.3-5.- Milestones for human expeditions to Phobos — prerequisite requirements.

2.1.4 Transportation Systems Definition

The transportation for Case Study 1 consists of the vehicle necessary to preplace cargo into Mars orbit, the vehicle to transport the astronauts to Mars, an optional vehicle for flights between the mother spaceship in Mars orbit and Phobos, a propulsion system for return of the piloted vehicle to Earth, and a capsule to permit direct descent of the crew to the Earth's surface. These vehicles are termed the Mars cargo vehicle (MCV), the Mars transfer vehicle (MTV), the Phobos excursion vehicle (PhEV), the trans-Earth injection system (TEIS), and the Earth crew capture vehicle (ECCV). These vehicles are summarized in table 2.1.4-I, which applies to Mars expeditions in general, including those discussed in section 2.2.4. This case study is baselined as all-propulsive, with only the ECCV using aeroassist.

2.1.4.1 Elements and Systems Description

Transportation Requirements/Assumptions. This mission consists of the MCV launch in February 2001, followed by the human mission launch in August 2002. A nominal mission time of 30 days in Mars orbit includes 20 days of human exploration of Phobos. Fortuitously, this launch opportunity allows a swingby of Venus, greatly reducing the need for propellants by eliminating an outbound deep space maneuver (DSM). Performing this mission at either the previous or next later launch opportunity greatly increases the propulsive mass requirements.

Other requirements and assumptions made for purposes of conducting the reference mission transportation analysis are given in table 2.1.4-II. These assumptions were necessary to achieve a point design for the reference mission. An analysis showed that initial mass in low Earth orbit (IMLEO) could be significantly reduced (IMLEO reduction of 59 percent) by using the relatively lightweight PhEV rather than requiring the entire manned spaceship to transfer to Phobos orbit because Phobos lies in a near-equatorial, circular orbit about Mars, necessitating major plane changes of the spacecraft's orbit. It was therefore found more effective to place the Mars orbiting vehicles (MOV's) into a high elliptical orbit of 250 km periapsis maneuvers. To achieve necessary rotation of the line of apsides, the orbit is circularized at 33,840 km altitude and then reinstated as an ellipse by an additional propulsive maneuver at an appropriate time before the final TEI burn. This special maneuver or some equivalent

TABLE 2.1.4-I.-HIERARCHIAL SUMMARY OF MARS VEHICLES AND FACILITIES

MSS	Mars Spaceship	The spaceship that is assembled in LEO
TMIS	Trans-Mars Injection System	Propulsion and guidance system for TMI
MTV	Mars Transfer Vehicle	Configuration during flight to Mars
IMM	Interplanetary Mission Modules	Hab/lab/log modules for crew in space
MOCS	Mars Orbital Capture System	Mars aerobrake + retro-propulsion + G&C
MCV	Mars Cargo Vehicle	Logistics vehicle sent for cargo staging
MDV	Mars Descent Vehicle	The vehicle which deorbits to land
MAV	Mars Ascent Vehicle	The vehicle which is launched to Mars orbit
MELS	Mars Entry & Landing System	Deorbit propulsion + aerobrake + parachute + terminal propulsion + G&C
MLMM	Mars Landed Mission Module(s)	Hab/lab/log modules
MLOE	Mars Landed Operations Equipment	Science, transportation, construction, manufacturing equipment — substitute S, T, C, M for O
RVR	Rover	Surface transportation for crew and/or exploration equipment.
MOV	Mars Orbiting Vehicle	Configuration in Mars orbit, not including the MDV's
TEIS	Trans-Earth Injection System	Propulsion and guidance system for TEI
PhEv	Phobos Excursion Vehicle	The piloted vehicle which leaves the MOV for rendezvous with Phobos/Deimos
ETV	Earth Transfer Vehicle	Configuration of the MSS for Mars-to-Earth transfer
MTM	Mars Transfer Modules	Hab/lab/log modules for crew in space
EOCS	Earth Orbital Capture System	Earth aerobrake + retro-propulsion, if required
ECCV	Earth Crew Capture Vehicle	Small vehicle for crew EOC and/or EELS
EELS	Earth Entry & Landing System	See MELS subsystems

TABLE 2.1.4-II.-TRANSPORTATION REQUIREMENTS AND ASSUMPTIONS

Requirements

- Man-rated transportation hardware 3 yr before launch; four 6-hr EVA's.
- Completion of microgravity countermeasures research (SRD p. 21)
- Minimum onorbit assembly and SS support; Split:sprint/conj.; flyaround aborts 1-2 yr in-LEO demo/verif of process requirements
- No radiation shielding for PhEV (from summary sheet), but required in MOV
- 20-30 days at Phobos, close proximity, "but not land per se."
- Docking capability with a previously planted anchor; "crew stability/mobility aids" for EVA work on Phobos
- Direct entry

Assumptions

- All-propulsive; ECCV for crew recovery at Earth; no recovery of ETV
- Excursion vehicle (PhEV, 9794 kg) to Phobos
- Crew contact with Phobos via EVA flight with MMU; PhEV does not contact Phobos surface.
- PhEV station - keeps 100 km from Phobos, with four 6-hr sorties to the surface
- Single TMIS stage, nonrecoverable
- Engine performance: Isp=480 for TMI, 460 for other cryo; 320 for storable biprop
- Propulsion: cryo for TMI, TEI, DSM, MOC, MOO; biprop for PhEV, RCS
- TMI Engine: single SSME (emergency use of MOCS for flyback in case of engine-out)
- Propulsion tankage factor: nominal (cryo: 0.15; storable: 0.058)
- Boiloff: low — 0.15 %/mo LEO, 0.3%/mo interplanetary (sprint), 0.065%/mo at Mars; high — 0.55 %/mo LEO, 1.0 %/mo interplanetary (sprint), 0.33%/mo at Mars
- Propellant margins: 1% each for DV, Isp, and bulk (use sum of margins) 2% mass margin on TEIS and ECCV retropropulsion (if required)
- Phobos science payload: 1.3 t, 200 w; two each 3.5 t MTR packages
- Hab modules: two SS-derived modules ("H" configuration)
- PVPA for spaceborne power, 200 m²
- Spaceborne ECLSS: closed for all, except food
- V: 100 m/s for MOO rendezvous, each vehicle, plus 619 m/s maneuvers to high circular orbit
- MCV drops MTR, RelayComSats from HEO-1 (prior to circ); PhEV from HEO-2

is required whenever short staytimes occur at Mars, such as for the sprint and opposition class trajectories, but can be accommodated during the longer conjunction class missions by orbital management strategies and much less expenditure of propulsion energy. Future studies will address alternative strategies for delta V reduction for short staytime missions.

The procedure for PhEV rendezvous with Phobos involves a sequence of plane change, altitude adjustment, and phasing burns. Upon rendezvous, a series of sorties to near the surface of Phobos allows EVA and exploration by one astronaut with the aid of a manned maneuvering unit (MMU). The other astronaut, fully suited, tends the depressurized PhEV and provides assistance, if needed, to the first astronaut. To return to the mother ship, the original sequence of orbital maneuvers is accomplished essentially in reverse.

Reference System Description.

Configuration and Mass Allocations. Cryopropellants (liquid hydrogen and liquid oxygen) are carried in standardized tanks designed to match the assumed ETO vehicle lift capability of 91 t to LEO. Tanks are 6.4 m diameter by 9.6 m long (21 ft dia x 31.6 ft), each holding 69.3 t of cryopropellant with an H/O ratio of 1:6. A pair of tanks is lifted each launch, as shown in figure 2.1.4-1. These "Siamese twin" tanks are not exactly identical but hold the same quantity of propellant and are connected by propellant transfer lines. The "wet" tank is ruggedized for the launch vibration, acceleration, and acoustic loads on a full tank, and equipped with foam insulation to store cryopropellant under atmospheric conditions prior to and during launch. Upon achieving orbit, propellant is

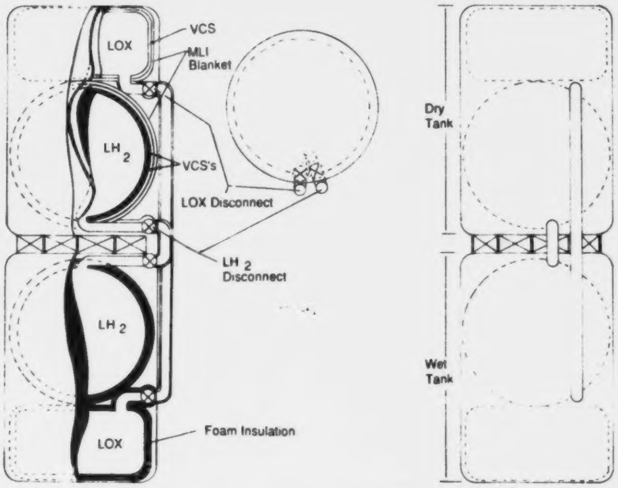


Figure 2.1.4-1.- Siamese twin tank concept.

transferred into the "dry" tank by an automated process. The dry tank is of lighter construction, with a 15 percent tankage factor (where tankage factor is the ratio of tank dry mass to propellant mass), and is thermally protected for long-term storage of cryogens in space by means of multilayer insulation blankets and vapor-cooled shields (VCS). This tank is to be used for all cryopropulsion stages, and achieves the low boiloff rates specified in the assumptions (table 2.1.4-II). The wet tanks are discarded after propellant transfer.

Conceivably they could be returned to Earth for reuse, or deorbited to prevent an accumulation of orbital debris. One or more tanks are pre-outfitted with an engine and propulsion avionics. To assemble a complete propulsion system, several tanks are docked together with their propellant fill ports connected. These ports are envisioned to be of technology derived from the STS 17-inch disconnects. During propulsive burn, all interconnected tanks drain propellant into the engine's primary tank.

The habitability module is shown in figure 2.1.4-2. It is an "H-module" configuration, consisting of two Space Station Freedom derivative modules with a single tunnel between them. At the midpoint of this tunnel, the ECCV is mounted. Because the ECCV ingress portal is in the nose, its interior is available to the crew at all times for habitation volume and access for continued training. It is necessary that the ECCV travel with the piloted vehicle so that it is available in case the optional Mars flyaround abort mode is selected in lieu of proceeding with Mars orbital capture and rendezvous with the cargo vehicle. No airlocks are provided, but EVA is made possible by venting the tunnel (which has an egress port opposite the ECCV) or one entire module.

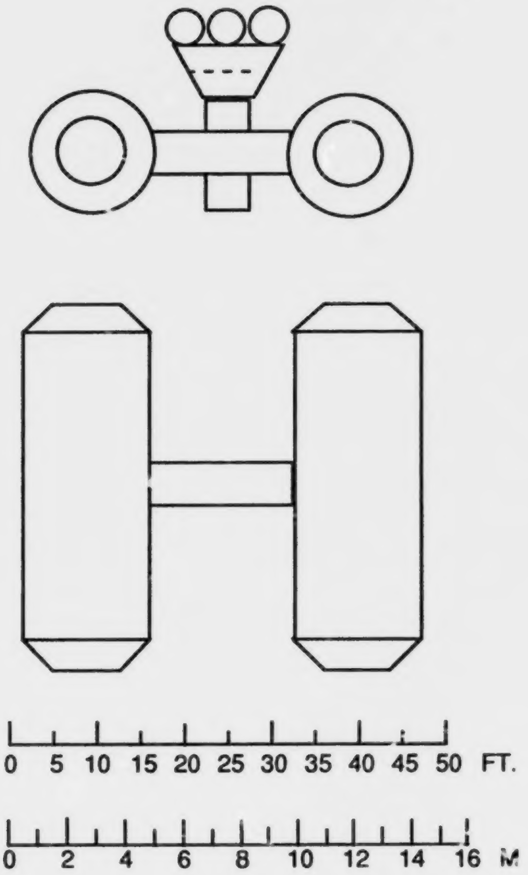


Figure 2.1.4-2.- H module.

The cylindrical habitats employ the pressure vessel and much support structure derived from Space Station Freedom modules. However, the massive internal experiment rack hardware is mostly replaced, both to provide a lighter-weight mounting structure appropriate to the small TMI propulsive loads and to allow more usable living volume for the astronauts. The interplanetary mission modules (IMM), including their electrical and communications support services, are sized at a mass of 44.3 t.

A solar flare radiation storm shelter is provided at the end of one of the modules. It consists of an approximately cubical volume designed to hold 4 persons (functional reach envelope for torso-restrained, unsuited 95th percentile male), as shown in figure 2.1.4-3. A minimum of 20 g/cm^2 shielding is provided in the walls of this shelter through judicious equipment installations, stowage of consumables and waste products, and added shielding and structure of 2.0 t. The shelter includes equipment for command and control of the spacecraft. Calculations taking into account astronaut mutual shielding and pathlength distributions for a mathematically idealized model of this shelter indicate a total integrated dose of less than 20 rem for each of the three worst known solar flare events (Feb 1956, Nov 1960, Aug 1972), and a total of less than 50 rem for the three events combined.

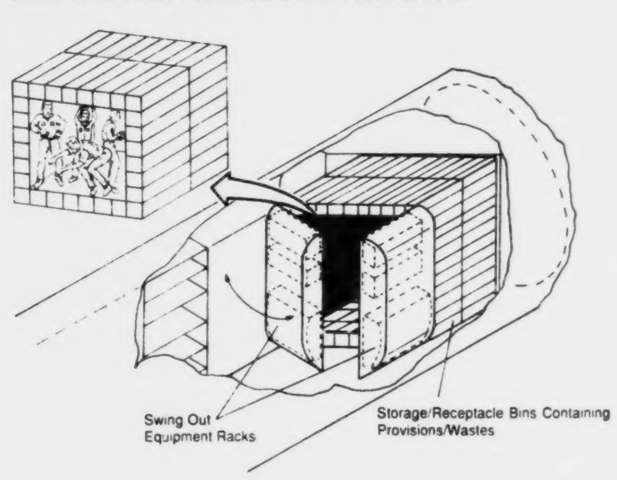

Swing Out Equipment Racks
Storage/Receptacle Bins Containing Provisions/Wastes

Figure 2.1.4-3.- Solar flare radiation storm shelter.

A life support system (LSS) with fourfold redundancy is provided at a mass cost of 2.8 t. The LSS provides recycling of oxygen, carbon dioxide, and water. A total 10.2 t of consumables (including 3.3 t food) is allocated to support the four astronauts for the duration of the mission, with 20 percent margin. Power is provided by two solar cell array wings of 100 m² each, deployed from the sides of the individual cylindrical modules.

The PhEV, figure 2.1.4-4, is similar to a Gemini capsule and holds a crew of two. Notable differences include capacity for a 700-kg Phobos science payload (or radiation shielding), two MMU units, and a major propulsion system. The propulsion employs the space-proven Delta engine and the associated storable bipropellants sized to accomplish a total round-trip delta V of 3327 m/s. The PhEV gross mass is just under 10 t.

The TEIS for Mars orbit escape consists of a single standard tank loaded with 51.9 t of cryopropellant. Three RL-10B-2 engines rated at a specific impulse of 460 s are provided. These engines are not used prior to the TEI burn. The engines are mounted in a close-packed triangular cluster, as indicated in figure 2.1.4-5, and the sys-

Dry Mass	
(includes payload)	3,757 kg
Payload Mass	
(user payload & crew)	1,070 kg
Payload Volume	
PhEV	
(cone - 1.8m rad., 2.3m ht.)	8 m
Propulsion System	
Propellant Type	MMH/NTO
Engines	
Number	1
Type	Delta
Mass (ea.)	100 kg
Thrust (total)	44.5 kN (10.0 f
I (320 sec)	klb)
Propellant Mass	3.14 kN - s/kg
Tank Mass	6,037 kg
Initial T/W	604 kg
Mass Fraction	1.24
Transfer to Phobos	
Phobos Sorties	0.7394
Transfer to 250 x 1 sol orbit	0.7365
Total Mass	0.6920
	9,794 kg

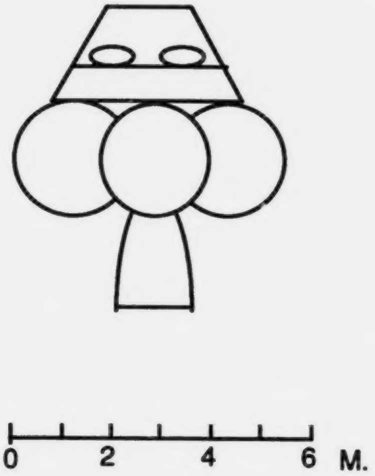
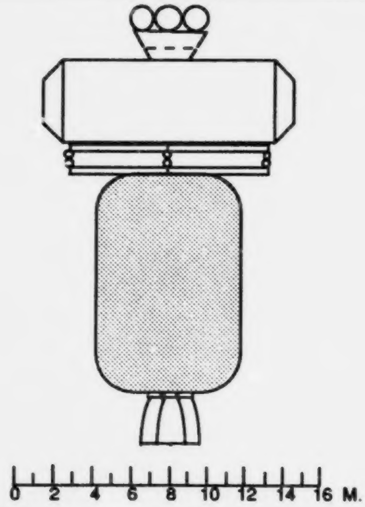


Figure 2.1.4-4.- Phobos excursion vehicle (PhEV).



(sprint class mission)

Dry Mass	
(includes payload)	67,960 kg
Payload Mass	
(ETV)	59,070 kg
Propulsion System	
Propellant Type	LOX/LH$_2$
Engines	
Number	3
Type	RL10B - 2
Mass (ea.)	191 kg
Thrust (total)	294 kN (66.1 klb$_f$)
I_{sp} (460 sec)	4.51 kN - s/kg
Propellant Mass	51,890 kg
Initial T/W	0.67
Mass Fraction	

Figure 2.1.4-5.- Earth transfer vehicle (ETV) with TEIS.

tem has one engine-out capability. Acceleration at TEI initiation is 0.18 g, rising to a peak acceleration of 0.32 g.

The ECCV can be likened to the Apollo Command Module. It enters the Earth's atmosphere at less than 12.2 km/s (40,000 f/s) with the aid of an aerobrake (for direct entry) and parachutes for terminal splashdown. The mass of the ECCV is 6.9 t, including the four crewmembers. No propulsion is required for the ECCV to accomplish its mission except a small propulsive system to accomplish final targeting (the main vehicle is targeted slightly off Earth intercept) and provide roll and attitude control.

The piloted vehicle stack in LEO is shown in figure 2.1.4-6, where the lower two tiers of tanks are associated with the trans-Mars injection system (TMIS). The propellant load of this system is 811.5 t, stored in twelve standard tanks. A Space Shuttle main engine (SSME)-derivative engine provides the thrust necessary for escape from LEO onto the interplanetary trajectory to Mars. The SSME-derivative employs an enlarged nozzle with 1000:1 expansion ratio and specific impulse of 480 s. The exit diameter of the bell is 8.2 m (27 ft). With its associated hardware, the nozzle + engine head + loaded tank + siamese twin tank (wet tank), the stack at launch is 28 m (92 ft) which may be accommodated depending upon the HLLV available. Alternatively, the nozzle could be segmented and then extended after launch, saving 8 m (26 ft) in stack length. The Mars orbital capture system (MOCS) is also a cryogenic propellant system, which in this case also provides Mars orbital operations (MOO) propellant sufficient for 1338 m/s of capability for apsidal rotation. It consists of four standard tanks, each with an RL-10B-2 engine. Engine-out capability is provided. Initial deceleration at the beginning of the MOC burn is 0.13 g.

The TMIS of the cargo vehicle, figure 2.1.4-7, consists of four cryopropellant tanks and one SSME-derivative

Dry Mass (includes payload)	499,850 kg
Payload Mass (MTV)	377,030 kg
Propulsion System	
Propellant type	LOX/LH $_2$
Engines	
Number	1
Type	SSME - derivative
Mass (ea.)	3,175 kg
Thrust (total)	2,4115 kN (543 klb $_f$)
I (480 sec)	4.71 kN - s/kg
Propellant Mass	811, 490 kg
Initial T/W	0.19
Mass Fraction	
Total	0.38
Total Mass	1,311,340 kg

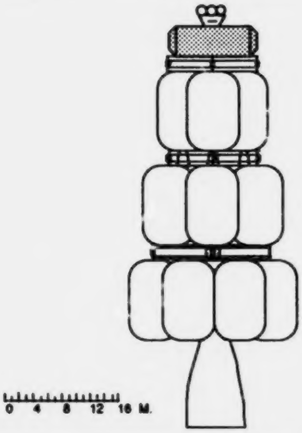
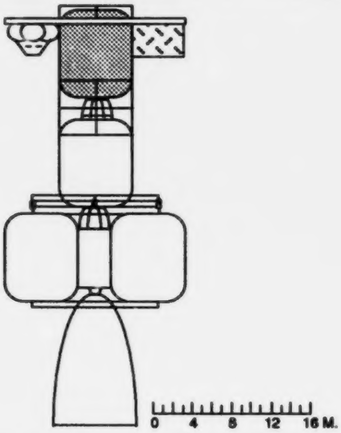


Figure 2.1.4-6.- Mars transfer vehicle (MTV) (piloted) with TMIS.



Dry Mass (includes Payload)	204,140 kg
Payload Mass (MCV)	163,610 kg
Propulsion System	
Propellant type	LOX/LH $_2$
Engines	
Number	1
Type	SSME - derivative
Mass (ea.)	3,175 kg
Thrust (total)	2,415 kN 543 klb $_f$)
I (480 sec)	4.71 kN - s/kg
Propellant Mass	262,830 kg
Initial T/W	0.53
Mass Fraction	
Total	0.56
Total Mass	466,970 kg

Figure 2.1.4-7.- Mars cargo vehicle (MCV) with TMIS.

engine. The MOCS/MOOS propulsion system is a single tank with a triangular cluster of RL-10B-2 engines. The cargo vehicle carries not only the TEIS and PhEV, but also a relay communication satellite and two Mars teleoperated rover (MTR) modules. The mass allocation for these additional payloads is 9 t, including all propellant loads necessary for these systems. An additional 0.45 t of instrument payload serves to provide solar flare monitoring and Mars orbital science capabilities.

Features of the System. This spacecraft is designed as a minimum system for accommodation of four astronauts for a deep-space mission. It provides somewhat more living volume per person than Space Station Freedom will provide to its occupants by virtue of the fact that not as much equipment will be installed. Neither airlocks nor nodes are provided. A cupola could be added to the end of one module at a small mass penalty. There is very little margin for error in rendezvous with the TEIS in Mars orbit. In the event that orbit insertion errors were large, the MCV could jettison the PhEV to lower its mass and hence provide some additional orbit modification capability in an effort to transfer to the orbit that the astronaut vehicle had reached.

ETO, Onorbit Assembly, and Servicing Needs. With an assumed HLLV capability of 91 t per launch, a minimum of 7 launches will be required to deploy the cargo vehicle and 18 launches for the piloted spacecraft and its propulsion systems. Of these, all but two will be dedicated solely to launch of propellant and propulsion system hardware. The launch profile is shown in table 2.1.4-III.

Onorbit assembly will be primarily by automated and teleoperated control from Earth. Propellant tank assembly into the necessary propulsion system clusters will be accomplished via docking maneuvers and plug-in propellant lines. An OMV, OMV/flight telerobotic servicer (FTS), and/or smart HLLV upper stage will be required as infrastructure to support this assembly. The IMM could be launched as a complete unit, including ECCV, if a 42-ft-diameter HLLV payload shroud were made available.

TABLE 2.1.4-III.- MINIMUM ETO YEARLY PROFILE FOR CASE STUDY 1 (NOT INCLUDING STS LAUNCHES)

Number of Launches	2000	2001	2002	Total
Baseline HLLV (91 t)	6	12	7	25
Very Large HLLV (200 t)		3	7	10
Magnum HLLV		1	1	2
Aerocapture (91 t HLLV)	4	5	3	12
Mass (t)				
Baseline HLLV (91 t)	468.3	1041.3	535.9	2045.5
Very Large HLLV (200 t)		591.4	1200.0	1791.4
Magnum HLLV		467.0	1311.3	1778.3
Aerocapture (91 t HLLV)	300.1	462.6	125.8	888.5

The "Siamese twin" tanks are designed for automatic propellant transfer in LEO, and no other propellant transfer capability is needed for this case study. All tanks are sized to allow for onorbit propellant boiloff losses. It is currently assumed that the MLI blankets and tankage walls will adequately minimize the probability of a leak induced by orbital debris and micrometeoroid impacts.

Other servicing requirements are also minimal and no STS visits are required for the cargo vehicle, although they may be desired for inspection of the assembled system. Onorbit operation and checkout of the IMM prior to final mating with the piloted TMIS will be required to develop the prerequisite baseline for mission assurance. A minimum of three STS launches is estimated to be required to support this mission.

Transportation Program Development Schedule. A schedule for development, proof-flight testing and man-rating of transportation hardware and propulsion systems is shown in figure 2.1.4-8. It must be stressed that the development of flight hardware must include time for prior development and man-rating of key elements in the mission. For example, the HLLV, PhEV, and TEIS characteristics will seriously affect design of both the cargo and piloted vehicles and therefore should be developed as early as possible. Demonstration of reliability can be made by precursor missions of various types, including Earth orbital and interplanetary unmanned launches.

One method of verification would be manned operation of the PhEV on a LEO mission which exercises the near-Phobos operational capabilities, followed by simulated rendezvous with the mother spacecraft. Additionally, the PhEV could also be operated unmanned to perform major orbital sequences at high altitudes to simulate the accuracy of transfers to and from Phobos. Both the HLLV and PhEV should be well into in-space testing by the mid-C/D phase of Mars vehicle developments, and preferably much earlier. Prototype habitability modules and other key components of the system could be pretested on Shuttle or Space Station Freedom flights. Long-term testing of life support system and microgravity countermeasures must be accomplished before the deadline for alteration of interior equipment complements for the piloted spacecraft. Long-term storage and successful operation of a TEI system must be tested in space prior to launch of a crew to Mars. If a Mars aerobrake is to be employed (see alternative, section 2.1.4.3), it should be assembled, onorbit launched, and entry-tested in the Earth's upper atmosphere prior to beginning the C/D phase of the piloted system. The need for additional

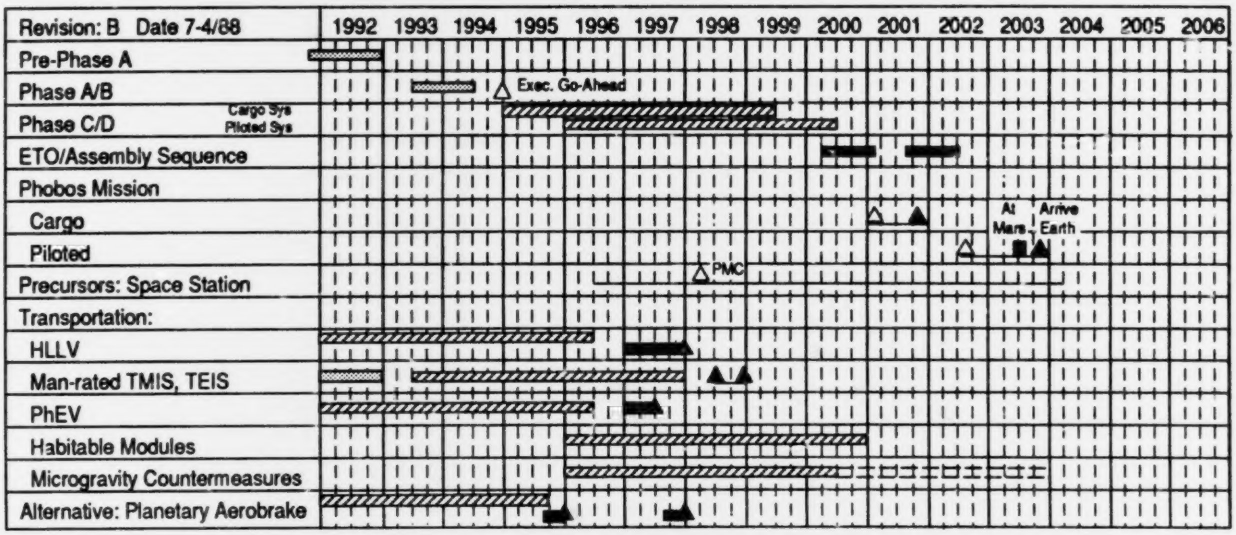


Figure 2.1.4-8.- Transportation program development schedule, Case Study 1.

aerobrake performance verification at Mars is under assessment.

Trades/Options. Several options have been studied for their effect on IMLEO of the total mass of cargo plus piloted vehicles, which is 1778.3 t for the baseline design. Use of conservative, high-boiloff tanks incurs a very significant additional mass penalty of 488 t. Conversely, if boiloff could be reduced to zero, some 152 t could be saved (less than a 9 percent reduction). A two-stage TMIS would save even less off the baseline design. Therefore, the more complicated system that would be required for successful staging is not adopted.

Advanced propulsion engines, with specific impulse capabilities of 485 s (TMIS) and 470 s for cryopropellant, and 340 for stored bipropellant, result in a savings of only 6 percent in IMLEO. In view of the development time lags associated with these advanced systems, they were not assumed for this case. Conservative tankage factors of 20.6 percent (compared to the 15 percent assumed) combined with high boiloff resulted in more than double IMLEO. Conversely, if the tankage factor could be reduced to 7.5 percent through use of advanced materials and technologies, more than 500 t reduction in IMLEO could be realized.

Major alternative approaches such as use of aerobraking at Mars and more advanced propulsion systems are considered briefly in section 2.1.4.3.

2.1.4.2 Enabling Technology Needs

Many technology needs are evident in missions of this class. First and foremost are development of an HLLV and of the LEO node, be it modifications to Space Station Freedom or a separate facility. A larger lift capacity of the HLLV will significantly reduce the number of launches and amount of onorbit assembly (see table 2.1.4-III). The shroud diameter is also of considerable importance. Space Station Freedom is required for studying the effects of and countermeasures against three major potential problems in long-duration space flight: deleterious adaptations to microgravity, diagnosis and treatment of complex medical problems, and psychosocial adjustment to the isolated and confined environment. Radiation hazards must also be understood and appropriate shielding provided. Other major developments include propulsion and storage of cryopropellants.

Propulsion Engines. Space-operated qualification of the SSME-derivative engine will be required. Increased performance of the SSME and RL-10 engines must be verified. Techniques for long-term in-space storage of the RL-10's must also be developed and tested.

Cryopropellant Tankage. It is quite obvious from the discussion in the previous **Trades/Options** paragraphs that every effort should be made for advancements in cryopropellant storage and for minimization of the tankage mass fraction relative to propellant (the "tankage factor"). This includes consideration of advanced composites, removable structures and shields, large multilayer insulation blankets, vapor cooled shields, and other options.

Precursor Missions. Selected missions will be needed to provide spaceborne demonstration/verification of the PhEV, ECCV, MTV, and propulsion systems.

2.1.4.3 System Alternatives and Opportunities

Use of an aerobrake to achieve Mars orbital capture allows elimination of the very large propulsion system otherwise required for the same purpose. For a Mars aerobrake (MAb) mass fraction of 10 percent, the reduction in total system mass is quite dramatic. IMLEO drops by a factor of nearly 2.4, to a value of 764.6 t. With this change, tankage and boiloff factors become relatively less important. For example, use of 7.5 percent tankage factor results in only an additional 20 percent IMLEO reduction. Even with high boiloff assumptions, and the adoption of a more storable TEI propellant such as hydrocarbon/liquid oxygen, the IMLEO increase is only about 10 t. This allows consideration of a more reliable TEIS, eliminating the difficulty of storage of liquid hydrogen for very long time periods.

Use of a nuclear thermal rocket (NTR) also results in major savings, based upon the demonstrated Nuclear Energy for Rocket Vehicle Applications (Nerva) technology which allows a specific impulse of 850 s. IMLEO is reduced by nearly 40 percent when nuclear propulsion is used for TMI, and by 52 percent when nuclear propulsion is used for all major maneuvers. A reduction of 71 percent can be obtained by combining Mars aerocapture and NTR technology.

2.1.5 Orbital Node Systems Definition

One of the major ground rules for this study was that no LEO transportation node would be used for the performance of the mission, and that Space Station Freedom support would be limited to life sciences research. Station modifications necessary for the life sciences support role are discussed in section 3.

2.1.6 Planetary Surface Systems Definition

Case Study 1 defines the first human expedition to the martian moon Phobos. Techniques will be developed to allow long-duration space flight and exploration of smaller orbiting bodies with miniscule gravity. The information gathered will be applied to future exploration of similar bodies and to establishing a human presence on Mars. Section 4.10 provides additional information about Phobos' physical characteristics and scientific exploration objectives.

2.1.6.1 Elements and Systems Description

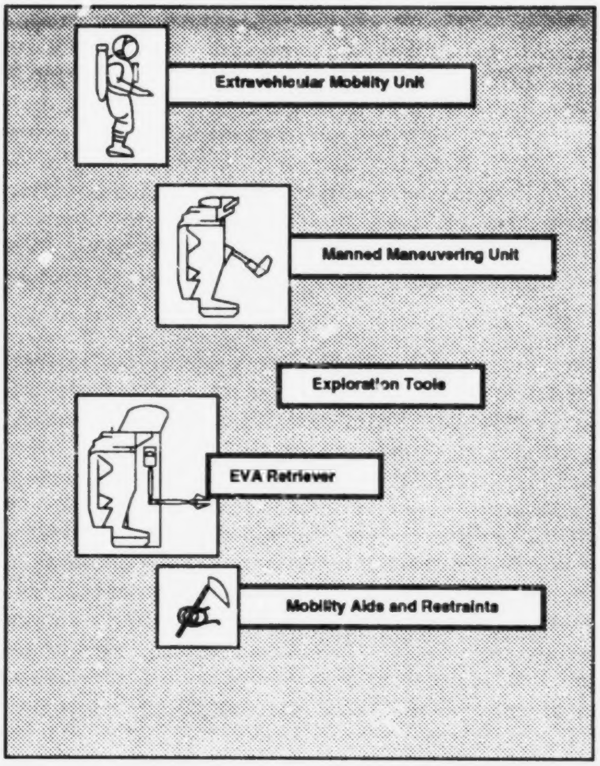


Figure 2.1.6-1.- Planetary surface systems elements.

Figure 2.1.6-1 summarizes the surface elements identified to support this case study. These include a space station extravehicular mobility unit (EMU), an enhanced version of the manned maneuvering unit (MMU) and its flight support system, an EVA retriever, a set of exploration tools (ET), and a set of mobility aids and restraints (MAR). In addition, a piloted orbital excursion vehicle is available as a habitat, science and experiments laboratory, source for replenishment of EVA consumables, and storage facility.

An allotment of up to two rovers exists as an adjunct to the Phobos exploration. Intended for teleoperation on the Mars surface by the crew in the Mars orbiting vehicle, each has the capability to return a sample to Mars orbit for analysis and/or return to Earth.

The mission to Phobos presents the first chance for humans to explore the surface of and assess the operations for exploration of an asteroid-type body. This body presents a unique set of environmental characteristics that must be considered in mission and contingency planning so that options can be preselected to ensure mission success. Specifically, the apparent inability to remain on the surface is a concern. There is a small gravity force which would keep a motionless body on the surface. However, the potential for leaving the surface for extended periods of time due to inadvertant pushes or bouncing off terrain features is high. This indicates a need for anchoring methods and propulsive systems to maintain or restore surface contact. Mountain climbing equipment such as tethers, chocks, pitons, camalots and ascenders may be appropriate. Some special anchoring equipment may

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need development. Surface characteristics of Phobos such as the softness of the regolith and its depth are unknown, as is the hardness of the rocky areas. Methods of anchoring people and equipment to this surface are not yet defined. As an adjunct, dust management is a concern. Speculations are that surface motions or jet plumes would not produce stable, localized dust clouds barring vision. However, dust could be hazardous to equipment during EVA and, if carried inside by returning EVA crews, to laboratory equipment and crew respiratory systems

Mobility on or above the surface is a concern. Gravity is sufficient to allow crewmembers some use of their arms and legs; however, hard shoves against the surface could produce long-duration flights (10 to 30 minutes) before recontact with the surface is reestablished. Although such excursions could be judiciously used, inadvertent flights could hinder crew operations and be hazardous. This may lead to the need for propulsive methods to regain or maintain surface contact. Crew use of tethers attached to pitons or camalots driven into the surface would ensure surface attachment during simple excursions, but special tools and equipment might still be required to provide stability for sampling activities such as boring into the surface or hammering. Surface vehicles that roll or are on tank treads present the same needs to regain or maintain surface contact. Mechanical hoppers are plausible, but increase the potential for mechanical failures and require attitude control systems. Flying vehicles provide the benefits of ease of motion over the surface and the ability to alight anywhere, but increase the use of propellants, generate rocket plumes, require GN&C systems, and may necessitate new development. If a vehicle is to be developed or modified, perhaps a flying vehicle which can be repeatedly anchored to and detached from the surface is the solution.

Flying over Phobos will be a unique experience. Figure 2.1.6-2 illustrates the motion of vehicle flight over the surface. The traveling distance is about 2 km. If the flight is to be a pogo emulation initiated by a 0.15 m/s delta V away from the surface, the flightpath may climb nearly 200 m and last slightly more than 3 hours. The initial delta V plus additional increments near the surface will require approximately 2.2 m/s total delta V. On the other hand, if the flight altitude is to remain at approximately 30 m and the velocity over the surface is to be about 1 m/s, the flight will last only approximately 30 minutes and require 3 m/s total delta V. The required velocity change required by these flights is an indication of potential vehicle requirements when integrated with the exploration requirements.

Case Study 1 involves 20 days on Phobos with a total of 24 hours of EVA (four 6-hour EVA's for each of two crewmembers). This EVA allotment, this personalized search and inspection, can be placed in a better perspective when viewed against the science objectives that could be proposed. Figure 2.1.6-3 shows some preliminary comments from the science community. First, the need for a global study of sites, iterative observations and sampling, and intensive human exploration is likely to far exceed the current EVA time allotment. This could

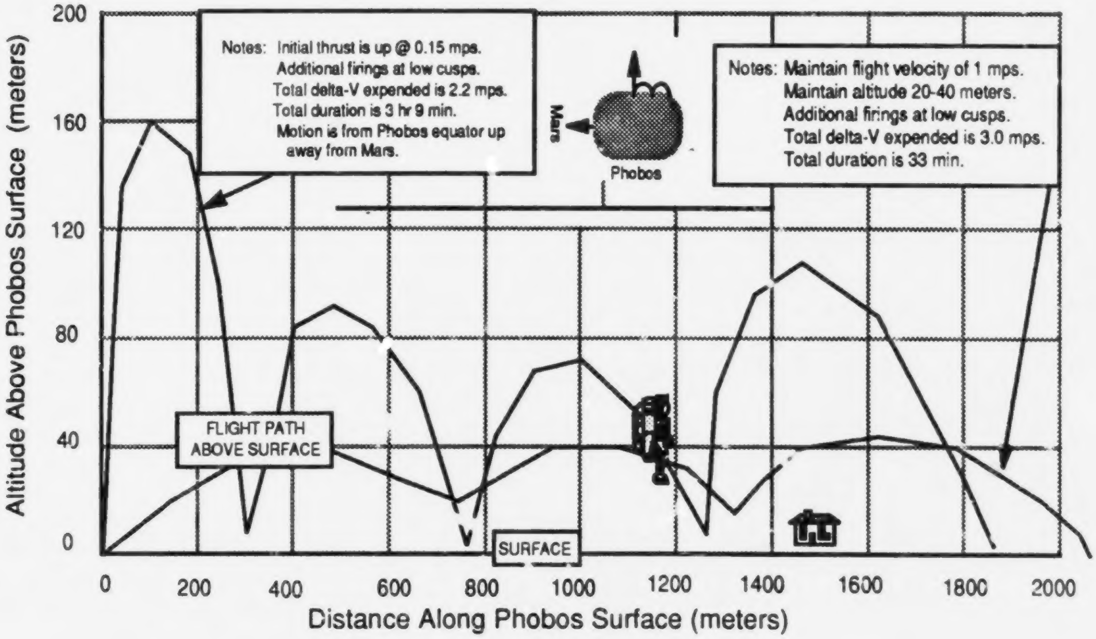

Figure 2.1.6-2.- Flying over Phobos.

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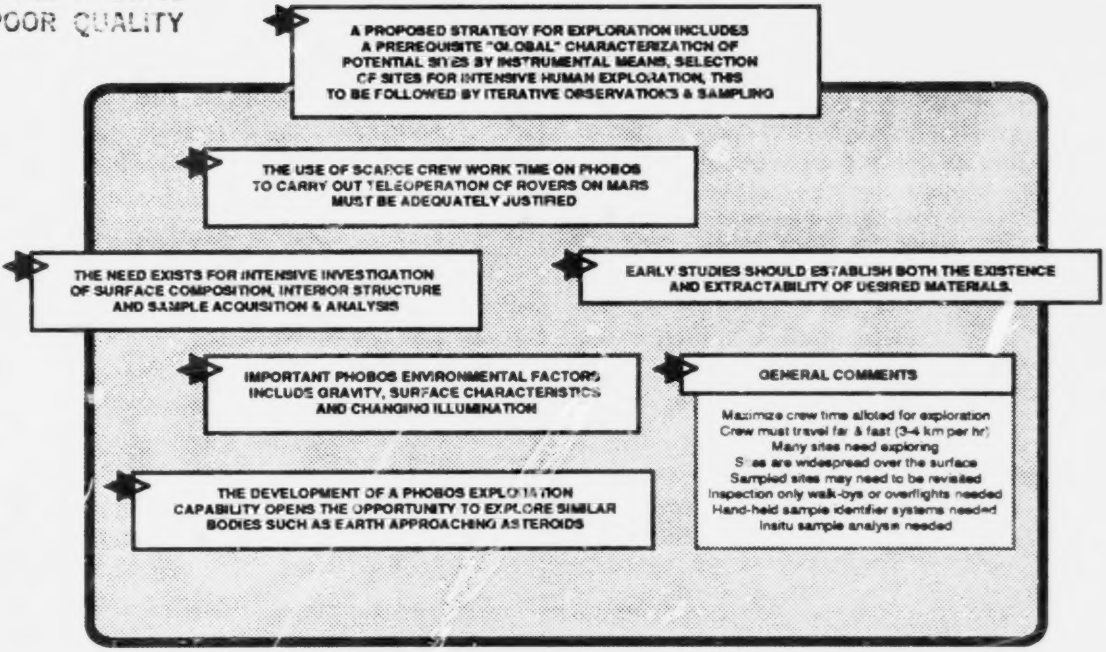

A PROPOSED STRATEGY FOR EXPLORATION INCLUDES A PREREQUISITE "GLOBAL" CHARACTERIZATION OF POTENTIAL SITES BY INSTRUMENTAL MEANS, SELECTION OF SITES FOR INTENSIVE HUMAN EXPLORATION, THIS TO BE FOLLOWED BY ITERATIVE OBSERVATIONS & SAMPLING

THE USE OF SCARCE CREW WORK TIME ON PHOBOS TO CARRY OUT TELEOPERATION OF ROVERS ON MARS MUST BE ADEQUATELY JUSTIFIED

THE NEED EXISTS FOR INTENSIVE INVESTIGATION OF SURFACE COMPOSITION, INTERIOR STRUCTURE AND SAMPLE ACQUISITION & ANALYSIS

EARLY STUDIES SHOULD ESTABLISH BOTH THE EXISTENCE AND EXTRACTABILITY OF DESIRED MATERIALS.

IMPORTANT PHOBOS ENVIRONMENTAL FACTORS INCLUDE GRAVITY, SURFACE CHARACTERISTICS AND CHANGING ILLUMINATION

GENERAL COMMENTS

Maximize crew time alloted for exploration
Crew must travel far & fast (3-4 km per hr)
Many sites need exploring
Sites are widespread over the surface
Sampled sites may need to be revisited
Inspection only walk-bys or overflights needed
Hand-held sample identifier systems needed
Insitu sample analysis needed

THE DEVELOPMENT OF A PHOBOS EXPLOITATION CAPABILITY OPENS THE OPPORTUNITY TO EXPLORE SIMILAR BODIES SUCH AS EARTH APPROACHING ASTEROIDS

Figure 2.1.6-3.- Preliminary comments from July 1988 science workshop.

create a requirement for the crew to travel relatively quickly (perhaps 3-4 km/h). An increase in systems capability is also appropriate. Crew aids such as the autonomous EVA retriever, which will provide additional safety for the crew, could also provide another (or other) explorer(s). These science objectives for global exploration and operations will be limited by the current EMU/MMU systems.

Detailed map preparation is needed as a precursor and/or an onsite task. Various sample collection activities will be required, some calling for in situ analysis. Verbal or written commentary on pure visual sightings, still photography, and video or movie filming are necessary. Long-term observations will be made by science packages left on the surface. Observation at various sites is a must. Precursor observations of Phobos from unmanned rovers may be required (at least on the immediately previous cargo flight). The same is true for Mars surface observations, which can be autonomous or teleoperated. These activities generate additional needs for mobility systems, vehicles, equipment, habitat, communications, navigation, guidance, control, mass, size, and logistics/resupply. Methods will be affected by the local environmental factors such as surface characteristics, surface elevation and slope, dust management, radiation, and gravity. Various types of equipment appear to be necessary for surface tasks. Anchors and penetrators are needed to create stable platforms. Solid extendable booms from the main vehicle are also a possibility. Tethers from the main vehicle or from stable platforms are an option. Cameras, sampling devices/tools, and portable carriers are a few others. Especially needed will be handheld detectors to aid the crew in the real-time selection of study sites and samples.

The rovers on Mars are intended to provide additional data about the planet while in the vicinity. Site selection must be consistent with descent and landing. Communication is necessary for teleoperating commands, video/data transfer and navigation. Also, ascent/rendezvous navigation must be provided to ensure an adequate trajectory. Teleoperation may also be conducted for a few days prior to Mars orbit insertion and after trans-Earth injection by the flightcrew. Limited teleoperation from Earth is a possibility.

2.1.6.2 Technology Drivers

There are several drivers to technology. However, these represent minor advances and are primarily in the area of equipment and systems development. They include mobility devices/vehicles, surface anchoring/stability systems, and sampling systems.

The teleoperation of rovers on Mars may also drive equipment and operations development. However, work here will be delayed into FY 1989 until the rovers' roles and tasks can be defined.

The need for surface travel and global sampling with minimal technological pressure suggests an already developed flying vehicle such as the EMU/MMU combination. This vehicle affords the pilot ample freedom for exploration. However, several enhancements are required to justify its use and increase its usefulness: additional development and modification to provide adequate safety and reliability of the unit itself during freeflight; development of the EVA retriever to aid in contingency rescue and crew exploration; guidance and navigation capability. The global exploration also implies an increase in available consumables for the extra "stay away" time and an increase in available delta V to allow longer and faster traverses.

The need to provide a stable work platform on the surface will require equipment development. As previously discussed, methods of anchoring to the surface must be devised, including a system to hold the EMU/MMU vehicle to the surface. Use of various types of mountain climbing equipment is possible, as well as drilling devices and pyrotechnic or explosive shell devices. Hard booms and platforms extending from the Phobos sortie vehicle are possible, although methods must still be developed to anchor this vehicle to the surface.

In addition, technology needs to provide for handheld sampling systems to help the crew recognize a desirable sample or location and to provide in situ analysis capability.

Although not a true precursor requirement by assumption, physiological requirements generated by long-duration space travel without gravity must be studied and adequately considered.

2.1.6.3 Systems Alternatives and Opportunities

Some mobility systems options, several of which have already been discussed, are illustrated in figure 2.1.6-4. Options are available for surface use or flight. The need for global exploration and the low gravity make the flight system the likely choice. The need for high crew freedom and minimal technological pressure leads to the EMU/MMU vehicle. However, staying with the Phobos sortie vehicle is possible. Use of foot and body restraints and hard extendable booms along with robotic arms is plausible. This method will severely limit crew mobility, require that the large vehicle use propellant to move about Phobos (or limit exploration further by not moving), and require the development and use of several movable arms.

In addition, the possibility exists for a surface roller or crawler. At 0.5 to 1.0 ft/s, assuming that movement takes place over 10 of the 20 days on Phobos, such a vehicle could traverse 70 to 140 miles of terrain. This vehicle would need an anchoring system compatible with vehicle movement while anchoring pins are in place. This means that some pins would need extraction, some would be set, while others were being set simultaneously. Problems occur here in that a totally surface-bound vehicle could be stopped by large rocks, holes, crevasses, steep slopes, extremely hard or soft soil, etc. This could lead to entrapment or at least detours that would take away traverse capability from exploration. However, no flight propellant would be required and perhaps power could be supplied by the Sun. There would be a need for extensive new vehicle development. Another option is to have this vehicle also capable of flight for fast traverse needs and for getting beyond obstacles.

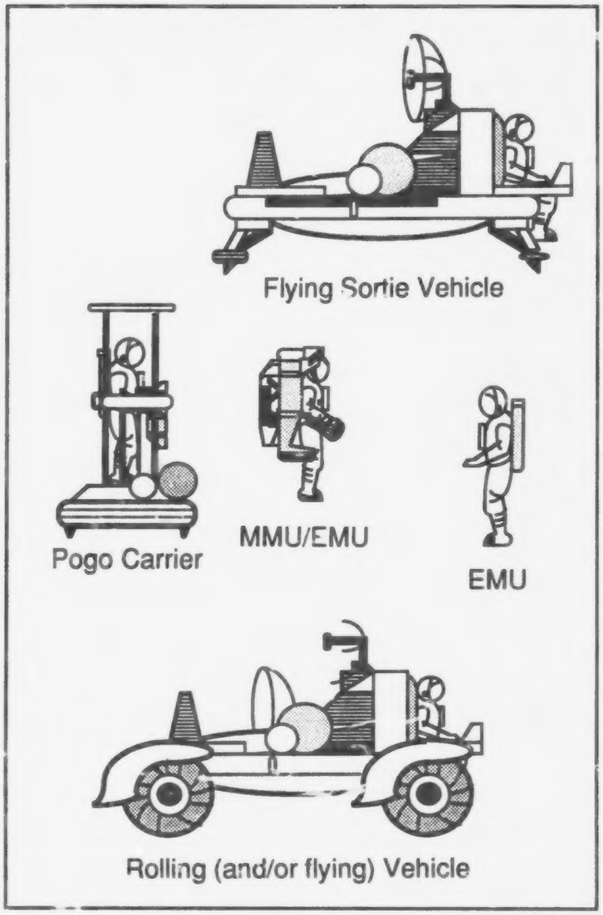

Figure 2.1.6-4.- Phobos exploration vehicle types.

There are several anchoring options. The need for a stable platform instead of a mere attachment leads away from tether only systems and towards self-effecting drills and/or pyrotechnic systems. Figure 2.1.6-5 shows an enhanced EMU/MMU vehicle equipped with an anchoring device is powered by explosive shells driving an anchor, attached to a cable, into the soil or rock. The vehicle reels the cable in, forcing a footpad to be securely pressed against the surface. Once stabilized, the crew can rotate the EMU/MMU to a desired azimuth over the terrain and pitch to a desired angle to the surface (including horizontal). The roll cage can be used to hang sampling and other equipment. Of course, stationkeeping flight is also possible, but it increases the use of propellant and complicates the vehicle control systems which must react to crew actions during sampling such as pulling, lifting, pushing, pounding, and drilling. Anchoring footpads could also be used by a "walking" EVA crewmember to allow adequate stability for exploration although they would limit the range of operations. Movement to a new site then must use the larger excursion vehicle. Various mountain climbing devices could be used.

The rovers on this mission are allotted to Mars surface exploration. One option generated by the science community is to use rovers to explore sites on Phobos prior to landing there. These could be the allotted rovers or additional rovers.

These options and possibilities were precipitated by the profound interest expressed by the science community. Answers to the questions abound and may be answered at the numerous exploration sites over the entire Phobos surface, each several kilometers in size and several kilometers apart.

2.1.7 Case Study Synthesis

2.1.7.1 Evaluation of Inputs

Consistency. There is a slight inconsistency associated with the PhEV payload capacity. The Transportation IA designed the PhEV for 900 kg of usable payload (700 kg for user payload, 200 kg for 2 spacesuits and MMU). The Planetary Surface Systems (PSS) IA planned for 1,260 kg of usable payload (680 kg for user systems and experiments and 578 kg for 2 spacesuits and MMUs). Thus, the PhEV is supposed to carry 360 kg more payload than it was designed for. The manifest in table 2.1.2-I uses the PSS payload mass numbers; figure 2.1.4-4 assumes the TA numbers. This inconsistency will not alter the basic design characteristics of the PhEV and is well within the margin of error expected for any space transportation vehicle point design at this early stage of analysis.

Parametric Results. For the most part, data were not submitted in parametric form. Although sufficient detail

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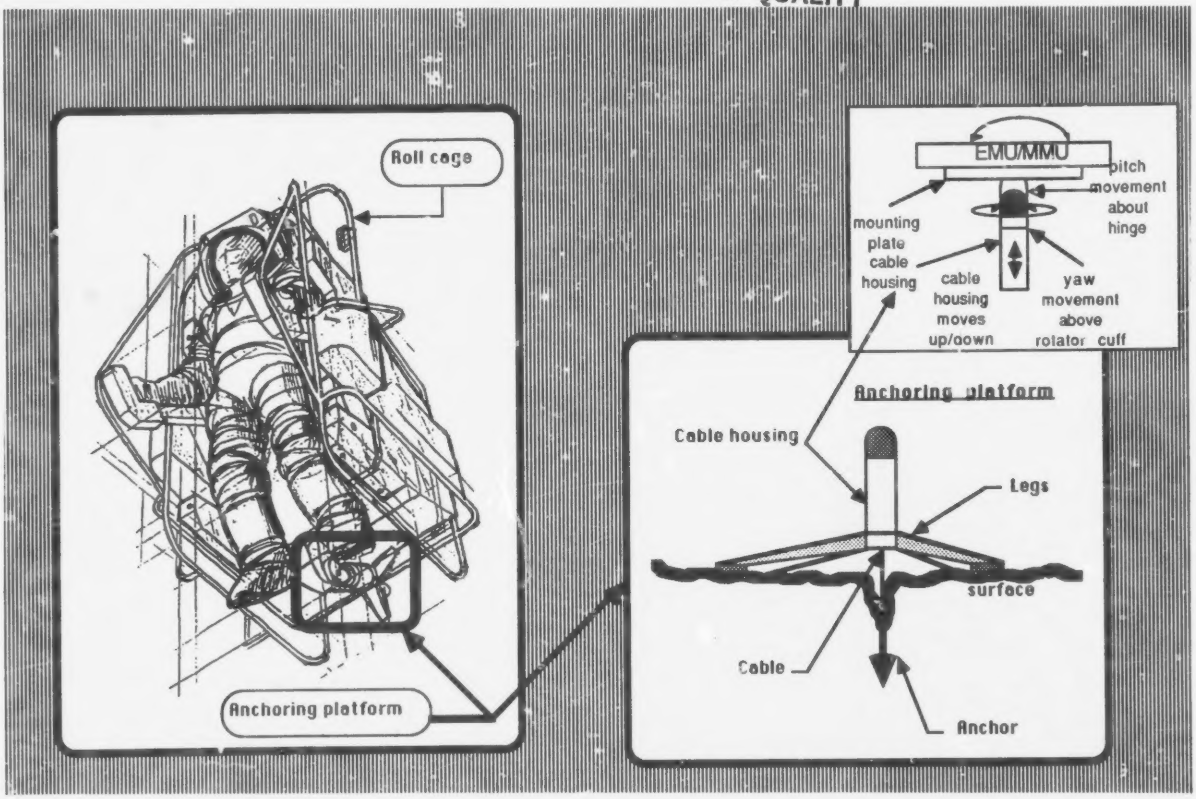

Figure 2.1.6-5.- Example of enhanced EMU/MMU with anchoring platform.

was provided in some areas for parametric relationships to be deduced, more explicit information is needed to support FY 1989 studies.

Options. The options explored by the Transportation IA in attempting to meet the LEO mass limitations were appropriate, and yielded useful data for the guidance of FY 1989 studies.

Special Assessments and Broad Trades. Many of the special assessments and broad trade results have only limited application to this case study because of its requirement to minimize reliance on new technology. Two exceptions are the special assessment of solid core nuclear thermal rockets (NTR's), and the broad trade study of end-to-end interplanetary spacecraft assembly in LEO. The latter was not completed in time to be useful to the Transportation IA, but the former was put to good use in exploring the option of using nuclear thermal in lieu of chemical rockets and thereby reducing the LEO departure mass of the Phobos expedition spacecraft.

2.1.7.2 Principal Issues and Program Risks

Zero-g Countermeasures. Probably the greatest program risk involved with the Phobos expedition plan (as described here) was recognized at the outset of the study: that the assumed success in developing zero-g countermeasures — which is the justification for omitting artificial gravity as a design feature of the crew carrier — may not materialize at all, or not as soon as needed.

Mars Aerocapture vs. NTR's. The LEO departure masses (table 2.1.1-II) for the baseline interplanetary spacecraft design are so great as to make the practicality of the mission doubtful in the absence of some kind of modification. Data developed by the Transportation IA show that the LEO mass could be reduced by something like 50 percent by using either of two technology alternatives, Mars aerocapture or NTR's. Both technologies used together would yield smaller LEO masses than either of them alone, but the additional benefit is unlikely to justify the added complexity and cost of doing so, at least for the Phobos expedition.

Solid-core NTR technology was developed to a near-operational status in the Nerva program. A total of 19 rocket reactors were built and tested to demonstrate NTR performance and restart capability. NTR advocates believe that outstanding problems are understood well enough that the work remaining to be done can be categorized as engineering rather than technology development. Most of the required testing can be done on the surface of the Earth, and the rest can be done in geosynchronous orbit.

Aerobrake enthusiasts opine that the development of Mars aerocapture technology is less risky than NTR development. Others believe that the development of Mars aerocapture technology will almost certainly require testing in the atmosphere of Mars itself, which will involve considerable time and logistical complexity. If so, given the need to compress the development schedule for this mission, the nuclear thermal option appears to be the better choice, but a thorough analysis needs to be made of safety considerations associated with ETO transportation of (virgin) reactors of the size required for manned Mars missions. Nerva program safety studies indicated that techniques exist to ensure ETO safety of NTR's.

ETO Payload Capacity of HLLV. Ever since the early 1960's it has been recognized that one of the major problems associated with manned Mars missions is that the masses of the spacecraft departing LEO are great enough to almost certainly require assembly in Earth orbit. It has also been recognized that the magnitude of the onorbit assembly problem decreases with increasing payload capacity (in terms of volume as well as mass) of the ETO launch vehicle. What remains to be decided is the optimum capacity, given especially that development of a launch vehicle design is very costly and should facilitate missions other than just manned ones to Mars.

2.2 HUMAN EXPEDITIONS TO MARS (CASE STUDY 2)

In this second case study, the prime objective is assumed to be, as before, the establishment of early leadership in manned exploration of the solar system. In the current instance this is referred to more accurately as a program objective rather than a mission objective, since the exploration plan calls for three separate human expeditions to be launched in successive Mars-mission opportunities.

2.2.1 Case Study Overview

2.2.1.1 Key Features

The surfaces of Mars, Phobos, and Deimos are explored by human extravehicular activity (EVA) during the opportunities that present themselves in the years 2007, 2009, and 2011. The three separate interplanetary flightcrews are each composed of eight members.

Baseline vehicle designs are again expendable and make no provision for artificial gravity, but they reflect a limited enhancement of technology for the specific purpose of enabling the Mars expeditions. In particular, orbital departure maneuvers at Mars and Earth are accomplished by advanced chemical rockets, and aerocapture technology is used both at Mars arrival and at Earth return.

2.2.1.2 Mission Profile

As in the case of the Phobos expedition, the split heliocentric trajectory profile described in section 2.1.1.2 was selected as nominal for these more ambitious Mars expeditions. For the crew carrier, the staytime in Mars orbit again was chosen to be 30 days. The round-trip time was limited to 440 days for the first two missions (2007 and 2009), but extended to 500 days for the third (2011). The extension was necessary to avoid a prohibitive low Earth orbit (LEO) mass penalty. An atmospheric entry speed limit of 8.62 km/s at Mars was imposed during heliocentric trajectory selection for both the cargo and the crew carriers, and a speed limit of 12.2 km/s for entry into the atmosphere upon the crew carrier's return to Earth.

Deep-space propulsive maneuvers are executed near perihelion to shape the outbound heliocentric legs of the crew-carrier trajectories in 2007 and 2009. In 2009 and 2011, Venus is in a favorable position to modify the heliocentric trajectory with its gravitational field, and is so employed.

Mars orbital operations are very much like those described in section 2.1.1.2, except that (1) the periapses of the approach hyperbolae lie within the atmosphere so that the aeroshields of the cargo and crew carrier vehicles can be used to provide the energy reduction required for capture into the initial areocentric ellipse, and (2) following rendezvous of the cargo and crew carriers, a lander/ascent vehicle is used to transport four of the crew to the surface of the planet and back again to the main spacecraft.

In addition to the trans-Earth injection (TEI) propulsion stage required for departure of the flightcrew from Mars orbit, the payload of the cargo carrier includes the to-be-manned Mars lander and ascent vehicle, teleoperable unmanned Mars surface rovers, and Mars orbital science and communication satellites. The cargo carrier for the first expedition also carries an excursion vehicle designed to transport two crewmembers to the surface of Deimos or Phobos, while two such vehicles are carried on the second cargo carrier, thus enabling human exploration of the surface of Mars and both of its moons during a single interplanetary expedition.

The landing crew spends 20 days on the surface of Mars, and in that time they make observations, conduct experiments, and gather samples with the aid of an unpressurized crew-carrying rover. Besides monitoring the activities of the surface exploration party, the four other crewmembers in the main spacecraft also teleoperate the unmanned rovers and — during the first and second mission opportunities — conduct their own explorations of Phobos and Deimos by means of the excursion vehicle designed for that purpose.

At the end of the interplanetary round trip, an aeroshield is used to brake a small module containing the flightcrew and the Mars samples into a geocentric orbit where they are transferred to a LEO node before eventually being ferried to the Earth surface.

2.2.1.3 Summary Data

Table 2.2.1-I contains a summary of major trajectory parameters for the nominal mission opportunities. Table 2.2.1-II contains corresponding LEO departure masses for the interplanetary vehicles. In the second table, mass data are shown for two of the more attractive alternatives to the baseline design. These are discussed in more detail in section 2.2.4.3.

In addition to the nominal opportunities, both of the aforementioned tables show data for the 2005 Mars exploration opportunity. The reason for this is that early in the study it was believed that the first manned landing on Mars could be made as early as 2005, and the corresponding trajectories were used in some of the important analyses and trade studies. Subsequent analysis of development schedules established that a manned landing is not a reasonable expectation before 2007; therefore the data for the 2005 opportunity should be used for reference only.

The planetary departure delta V values shown in table 2.2.1-I represent impulsive velocity increments for departure from optimally oriented planetocentric orbits. The predeparture orbit altitude at Earth was assumed to be 500 by 500 km. The departure increments at Mars represent the periapsidal components of Oberth escape maneuvers from a circular orbit having an altitude of 33,840 km. They do not include the 619 m/s required to establish an elliptical periapse at 250 km before application of the final TEI impulse. In the computation of atmospheric entry speeds, the entry altitude was taken to be 121.9 km at Mars and at Earth.

The qualifying remarks in the last two paragraphs of section 2.1.1.3 (relating to orbital departure windows, gravity losses, performance reserves, and optimality of the heliocentric trajectory selections) apply equally as well to the data in tables 2.2.1-I and 2.2.1-II as they do to the data in tables 2.1.1-I and 2.1.1-II.

TABLE 2.2.1-I.- TRAJECTORY DATA FOR MARS EXPEDITION

Vehicle	Event/Parameter	Mission Opportunity			
		* 2005†	2007†	2009†	2011
Cargo Carrier	**LEO Departure**				
	Elapsed Days	0	0	0	0
	Date	2003 Jun 07	2005 Sep 01	2007 Sep 22	2009 Oct 14
	Decl Vinf (deg)	-7	12	18	22
	Delta V (m/s)	3555	3846	3730	3620
	Mars Arrival				
	Elapsed Days	202	402	369	329
	Date	2003 Dec 25	2006 Oct 08	2008 Sep 25	2010 Sept 08
	Decl Vinf (deg)	7	26	14	-3
	Entry V (m/s)	5627	6053	5687	5520
Crew Carrier	**LEO Departure**				
	Elapsed Days	0	0	0	0
	Date	2004 Oct 09	2006 Dec 31	2009 Feb 06	2011 Jan 06
	Decl Vinf (deg)	-6	-18	17	-4
	Delta V (m/s)	4556	6068	5692	6190
	Venus Swingby				
	Elapsed Days			52	126
	Date			2009 Mar 30	2011 May 12
	Solar dist (au)			0.72	0.73
	Min Alt (km)			1745	1976
	Deep-Space Mnvr				
	Elapsed Days	99	81	91	
	Date	2005 Jan 17	2007 Mar 22	2009 May 08	
	Solar dist (au)	0.66	0.63	0.56	
	Delta V (m/s)	1897	1775	1173	
	Mars Arrival				
	Elapsed Days	266	229	238	264
	Date	2005 Jul 02	2007 Aug 17	2009 Oct 02	2011 Sep 27
	Decl Vinf (deg)	1	17	30	25
	Entry V (m/s)	8227	8419	8540	7803
	Mars Departure				
	Elapsed Days	296	259	268	294
	Date	2005 Aug 01	2007 Sep 16	2009 Nov 01	2011 Oct 27
	Decl Vinf (deg)	-23	-13	7	22
	Delta V (m/s)	2044	2652	3766	1820
	LEO Return				
	Elapsed Days	440	440	440	500
	Date	2005 Dec 23	2008 Mar 15	2010 Apr 22	2012 May 20
	Decl Vinf (deg)	-11	-15	-20	-6
	Entry V (m/s)	12151	11606	11889	12151

* For reference only
† Crew arrival dates at Mars

TABLE 2.2.1-II.- LEO DEPARTURE MASS FOR MARS EXPEDITION

Design Option	Mission Opportunity				LEO Departure Mass Component
	* 2005†	2007†	2009†	2011†	
Baseline	503 t 1125 t 1628 t	747t 1770 t 2517 t	1123 t 1512 t 2635 t	629 t 1134 t 1763 t	Cargo Carrier Crew Carrier Total
Nuclear Thermal Rockets	328 t 707 t 1035 t	———	———	———	Cargo Carrier Crew Carrier Total
Conjunction Class Chemical	1069 t	1126 t	977 t	1018 t	All-up Cargo + Crew Carrier w Artificial g (980 Day Total)

* For reference only
† Dates at top of columns are crew arrival dates at Mars

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2.2.2 Mission Definition and Manifest

Precursors to the Mars expeditions begin with the launch of the Mars Observer satellite mission in 1992, followed by the Mars Rover Sample Return mission beginning in 1994. These unmanned missions are used to acquire precursor scientific data on the Mars system needed for planning the human expeditions to Mars.

The buildup of LEO supporting infrastructure begins in 1999 with the implementation of a variable-g facility at or near Space Station Freedom. The LEO infrastructure buildup continues with the construction of a vehicle assembly and propellant depot structure which is completed in 2003. The first of three interplanetary cargo vehicles is launched from LEO in 2005, followed by departure of the first interplanetary crew carrier in 2006.

Table 2.2.2-I contains a list of payload items and the quantity of each item to be delivered to its appropriate destination (LEO for node system elements, Mars for all others) during each year of the 20-year program assumed for this case study.

TABLE 2.2.2-I.- HUMAN EXPEDITIONS TO MARS PAYLOAD ELEMENT MANIFEST

Mars Orbit Systems Science Element Name	Mass kg	LEO Departure Dates													
		99	00	01	02	03	04	05	06	07	08	09	10	11	12
Sample Collection Equipment Set	100.00							1		2					
Phobos/Deimos Simple Seismic Network	300.00							1		1					
Teleoperated Geophysical/Meteorological Lander	2000.00							2		1		3			
Teleoperated Sample Collection Lander	4000.00									2		1			

Mars Orbit Systems Support Element Name	Mass kg	LEO Departure Dates													
		99	00	01	02	03	04	05	06	07	08	09	10	11	12
Space Suit (EMU)	135.90							2		4					
MMU	153.11							2		4					
MMU Flight Support System	114.76							1		2					
EVA Retriever	480.18							1		2					
Phobos/Deimos Mobility Aids and Restraints	100.00							1		2					
Areosynchronous Communication Satellites	500.00							3							

Mars Surface Systems Science Element Name	Mass kg	LEO Departure Dates													
		99	00	01	02	03	04	05	06	07	08	09	10	11	12
Unpressurized Manned Rover	1000.00							1		1		1			
Robotic Rover	1000.00							2		2		2			
Meteorological Balloon	1000.00							1		1		1			
Biology Experiment Package	300.00							1		1		1			
Sample Collection Equipment Set	100.00							2		2		2			
Geophysical/Meteorological Station	160.00							1		1		1			
Portable Traverse Geophysical Experiment Pckg.	100.00							1		1		1			
Subsurface Core Drill	1200.00							1		1		1			
Drill Power Supply	3500.00							1		1		1			

Mars Surface Systems Support Element Name	Mass kg	LEO Departure Dates													
		99	00	01	02	03	04	05	06	07	08	09	10	11	12
Mars Suit	229.22							4		4		4			
Radiation Protection Garment	170.00							2		2		2			
Regolith Bagger	250.00							1		1		1			

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TABLE 2.2.2-I.- Concluded

Transportation Systems Element Name	Mass t	LEO Departure Dates													
		99	00	01	02	03	04	05	06	07	08	09	10	11	12
Cargo Mission #1: TMI Stage (dry)	64.50							1							
TMI Cryo. Props. (includes MOC, MOO)	455.62							1							
Storable Props. (includes MCC, RCS)	15.17							1							
Cargo Mother Ship	33.95							1							
TEI Stage (dry)	14.86							1							
TEI Cryo. Props.	91.75							1							
Cargo Mission #2: TMI Stage (dry)	94.34									1					
TMI Cryo. Props. (includes MOC, MOO)	673.65									1					
Storable Props. (includes MCC, RCS)	23.82									1					
Cargo Mother Ship	48.78									1					
TEI Stage (dry)	26.78									1					
TEI Cryo. Props.	171.19									1					
Cargo Mission #3: TMI Stage (dry)	94.81											1			
TMI Cryo. Props. (includes MOC, MOO)	345.57											1			
Storable Props. (includes MCC, RCS)	5.47											1			
Cargo Mother Ship	68.74											1			
TEI Stage (dry)	-											1			
TEI Cryo. Props.	48.83											1			
Crew Mission #1: TMI Stage(s) (dry)	187.60								1						
TMI Cryo Props. (includes DSM, MOC, MOO1)	1375.30								1						
Storable Props. (incudes MCC, RCS, EOC)	14.28								1						
Crew Mother Ship	151.80								1						
TEI Preparation Cryo. Prop. (MOO2)	40.70								1						
Crew Mission #2: TMI Stage(s) (dry)	156.26											1			
TMI Cryo Props. (includes DSM, MOC, MOO1)	1132.00											1			
Storable Props. (incudes MCC, RCS, EOC)	15.02											1			
Crew Mother Ship	152.30											1			
TEI Preparation Cryo. Prop. (MOO2)	56.24											1			
Crew Mission #3: TMI Stage(s) (dry)	147.12													1	
TMI Cryo Props. (includes DSM, MOC, MOO1)	797.97													1	
Storable Props. (incudes MCC, RCS, EOC)	15.43													1	
Crew Mother Ship	143.17													1	
TEI Preparation Cryo. Prop. (MOO2)	30.73													1	
Mars Descent Vehicle (dry, minus payload)	18.36							1		1		1			
Mars Descent Vehicle Propellant	3.41							1		1		1			
Mars Ascent Vehicle (dry, minus payload)	3.29							1		1		1			
Mars Ascent Vehicle Propellant	19.42							1		1		1			
Phobos/Deimos Excursion Vehicle (dry)	2.56							1		2					
Phobos/Deimos Excursion Vehicle Propellant	5.57							1		2					

Nodes Systems Element Name	Mass t	LEO Operational Date													
		99	00	01	02	03	04	05	06	07	08	09	10	11	12
LEO Mars Vehicle Assembly/Propellant Depot	121.70					1									

2.2.3 Mission Architecture and Infrastructure

The mission architecture for this case study is illustrated in figures 2.2.3-1 and 2.2.3-2. The STS Shuttle is used for Earth-to-orbit (ETO) transportation of flightcrews, and heavy-lift launch vehicles (HLLV's) are used for all other ETO transportation.

Transportation requirements for the program of Mars expeditions are shown graphically in figure 2.2.3-3. Maximum ETO mass in any one year is about 1800 metric tons in 2006, for the most part consisting of the LEO departure mass of the interplanetary crew carrier that arrives at Mars in 2007.

Variation of the LEO departure masses of the split/sprint cargo and crew carriers from one Mars mission opportunity to another is due primarily to the eccentricity of Mars' orbit about the Sun. This variation is complicated by the asynchronism of the motions of Mars and Venus relative to Earth, which results in the availability of an energy-saving Venus swingby during some opportunities, but not in others.

Major milestones for the program of human expeditions to Mars are shown in figures 2.2.3-4 and 2.2.3-5.

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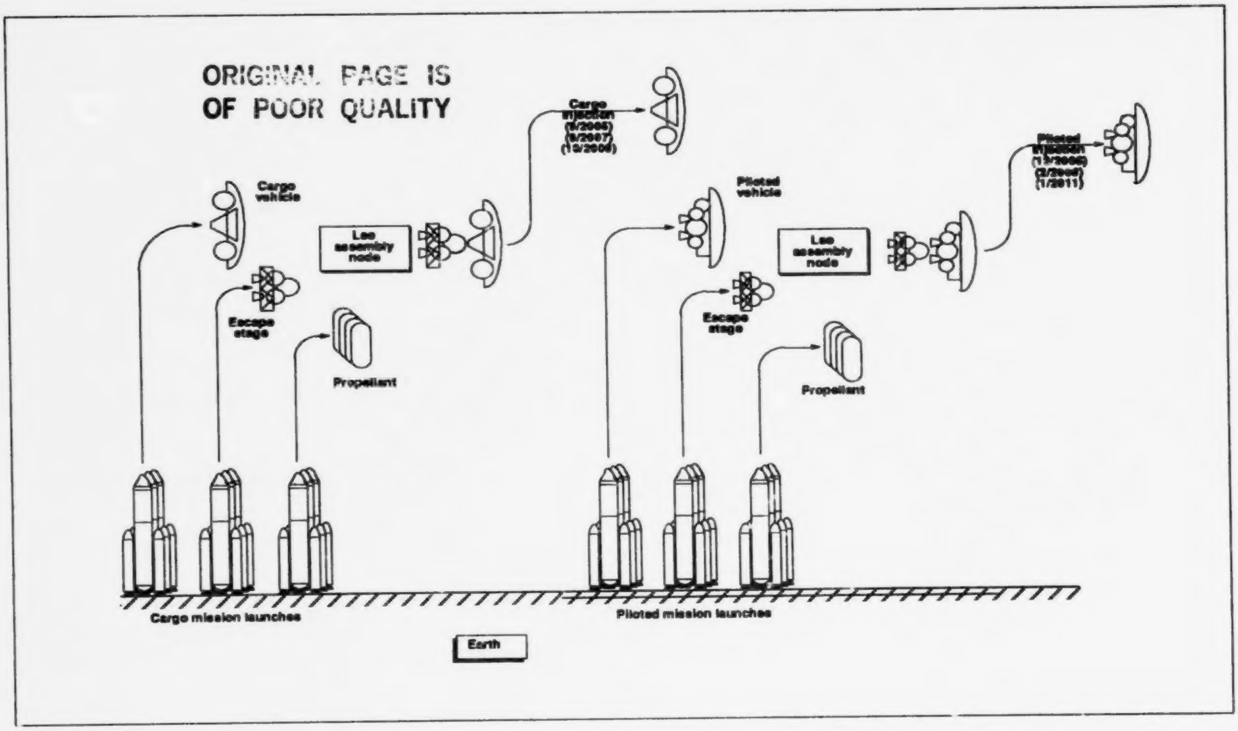

Cargo vehicle
Escape stage
Propellant
Leo assembly node
Cargo injection
(5/2005)
(9/2007)
(10/2009)
Piloted vehicle
Escape stage
Propellant
Leo assembly node
Piloted injection
(11/2006)
(2/2009)
(1/2011)
Cargo mission launches
Piloted mission launches
Earth

Figure 2.2.3-1.- Human expeditions to Mars — Earth orbital operations.

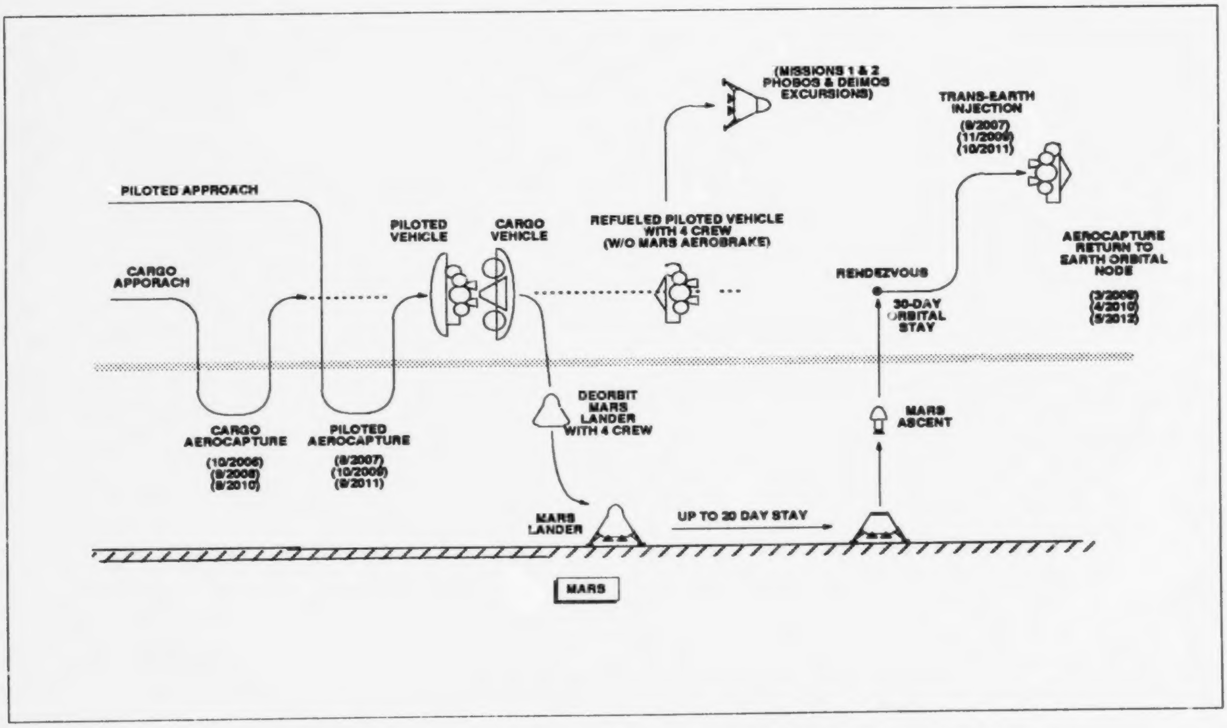
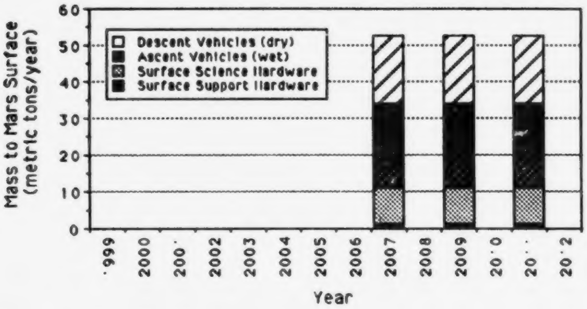

(MISSIONS 1 & 2
PHOBOS & DEIMOS
EXCURSIONS)
TRANS-EARTH
INJECTION
(9/2007)
(11/2009)
(10/2011)
PILOTED APPROACH
CARGO
APPORACH
PILOTED
VEHICLE
CARGO
VEHICLE
REFUELED PILOTED VEHICLE
WITH 4 CREW
(W/O MARS AEROBRAKE)
RENDEZVOUS
30-DAY
ORBITAL
STAY
AEROCAPTURE
RETURN TO
EARTH ORBITAL
NODE
(3/2008)
(4/2010)
(5/2012)
CARGO
AEROCAPTURE
(10/2006)
(9/2008)
(8/2010)
PILOTED
AEROCAPTURE
(8/2007)
(10/2009)
(9/2011)
DEORBIT
MARS
LANDER
WITH 4 CREW
MARS
ASCENT
MARS
LANDER
UP TO 20 DAY STAY
MARS

Figure 2.2.3-2.- Human expeditions to Mars — Mars orbital/surface operations.

Human Expeditions to Mars



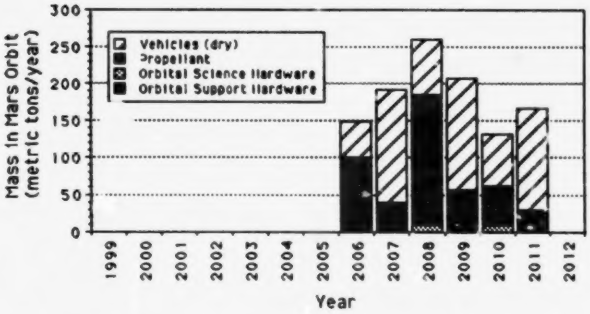
Notes: Mass to Mars Surface

Descent Vehicles (dry) includes the dry descent stage of the crew lander/habitation vehicles

Ascent Vehicles (wet) includes the ascent stage of the crew lander/habitation vehicle and the storable propellant needed to launch it and the crew from the martian surface to Mars orbit for rendezvous with the waiting mother ship

Surface Science Hardware includes the equipment used to conduct the science experiments on Mars (e.g., geophysical/ meteorological stations, meteorological balloons, biology experiment packages)

Surface Support Hardware includes the equipment used to support the science experiments on Mars (e.g., space suits, radiation protection garments, regolith bagger)



Notes: Mass in Mars Orbit (this includes things used in Mars orbit)

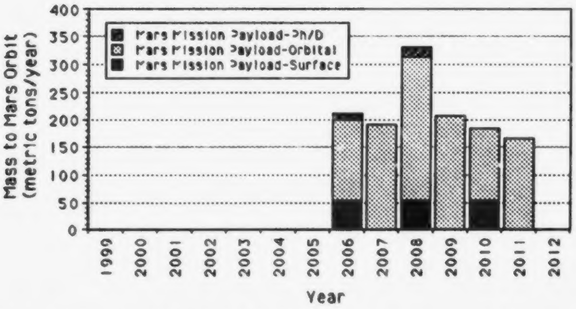
Vehicles (dry) includes the dry trans-Earth injection stages, the dry crew mother ships, and the dry cargo mother ships

Propellant includes the storable descent propellant in the Mars crew lander/habitation vehicles and the cryogenic propellant used to return to Earth

Orbital Science Hardware includes the equipment used to conduct the science experiments in Mars orbit (e.g., teleoperated landers, Mars science satellites, etc.)

Orbital Support Hardware includes the equipment used to support the science experiments in Mars orbit (e.g., Mars communications satellites)

Note: In missions 1 and 2 the complete TEI stage (wet) is carried to Mars by the cargo vehicles. In mission 3 the TEI propellant is carried to Mars by the cargo vehicle and the dry TEI stage is carried to Mars by the crew vehicle



Note: Mass to Mars Orbit (This includes everything taken to Mars)

Mars Mission Payload-Ph/D includes the wet Phobos and Deimos excursion vehicles and all science and support hardware taken to Phobos and Deimos

Mars Mission Payload-Orbital includes the wet trans-Earth injection stages, the crew and cargo mother ships, the storable descent propellant for the Mars crew lander/habitation vehicles, and all science and support hardware used or deployed in orbit

Mars Mission Payload-Surface includes the dry descent stage and the wet ascent stage of the Mars crew lander/habitation vehicles and all science and support hardware used on the martian surface

Note: In missions 1 and 2 the complete TEI stage (wet) is carried to Mars by the cargo vehicles. In mission 3 the TEI propellant is carried to Mars by the cargo vehicle and the dry TEI stage is carried to Mars by the crew vehicle

Figure 2.2.3-3.- Human Expeditions to Mars — transportation requirements.

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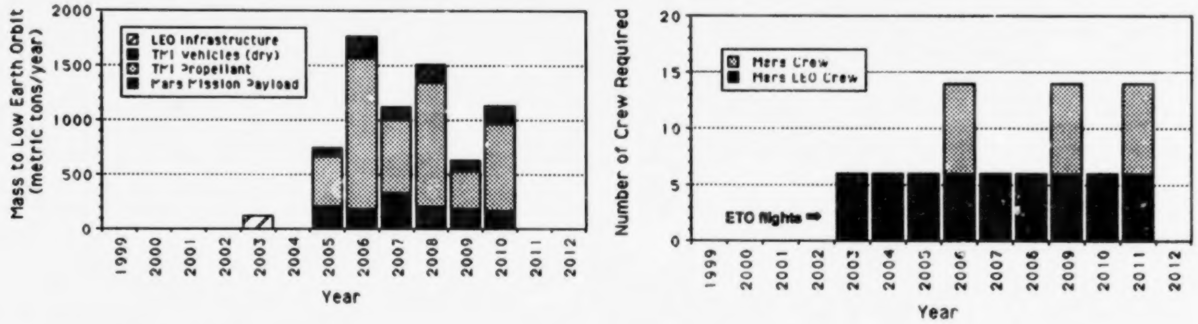
Human Expeditions to Mars



Notes: Mass to Low Earth Orbit

LEO Infrastructure includes a Mars vehicle assembly/propellant depot

TMI Vehicles (dry) includes the dry trans-Mars injection stages and the storable propellants and crew consumables used in getting to Mars

TMI Propellant includes the cryogenic propellant used to get the cargo and crew vehicles to Mars

Mars Mission Payload includes the crew and cargo motherships, the cryogenic propellant used to return the crew to Earth, the Mars crew lander/habitation vehicles, the Phobos and Deimos excursion vehicles, and all science and support hardware used at Phobos and Deimos, in Mars orbit, and on the martian surface

Note: In missions 1 and 2 the complete TEI stage (wet) is carried to Mars by the cargo vehicles. In mission 3 the TEI propellant is carried to Mars by the cargo vehicle and the dry TEI stage is carried to Mars by the crew vehicle

Note: Number of Crew Required

Mars Crew is the eight-person crew sent to Mars

Mars LEO Crew is the crew used to construct an assembly/propellant depot and the trans-Mars Vehicles

ETO flights based on 180 day shifts and personnel carrier capacity of 11

Figure 2.2.3-3.-(Concluded).

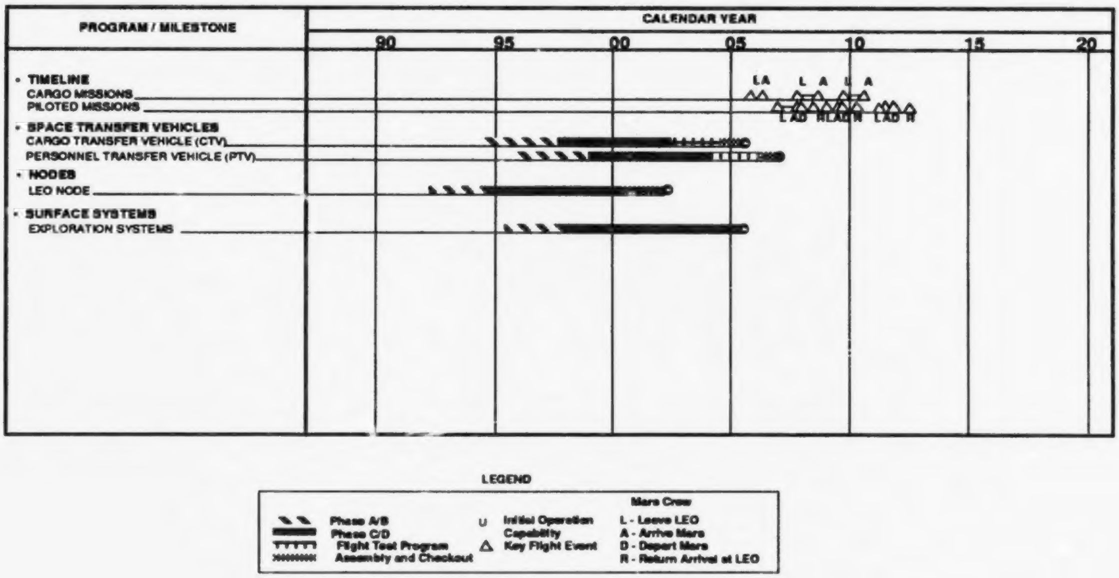


Figure 2.2.3-4.- Milestones for human expeditions to Mars — mission requirements.

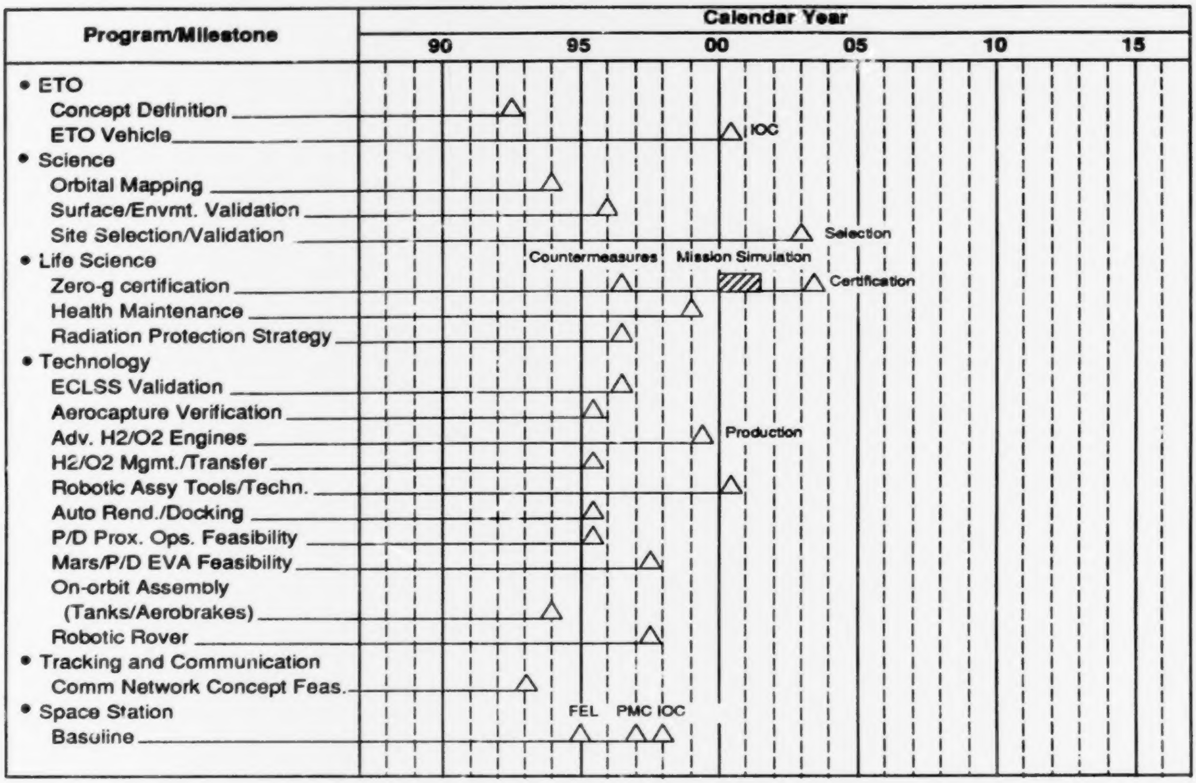


Figure 2.2.3-5.- Milestones for human expeditions to Mars —prerequisite requirements.

2.2.4 Transportation Systems Definition

The transportation for Case Study 2 consists of a series of three split missions to Mars, including Mars cargo vehicles (MCV's), Mars spaceships (MSS), Mars descent vehicles (MDV's), Phobos excursion vehicles (PhEV's), and a Deimos excursion vehicle (DeEV). Each MDV includes within it a Mars ascent vehicle (MAV) for return of the landed astronauts to the Mars orbiting vehicle (MOV). Also used in this case study are TEIS and aerocapture ECCV's. A hierarchial summary of Mars vehicles and facilities is contained in table 2.1.4-I.

2.2.4.1 Elements and Systems Description

Transportation Requirements/Assumptions. These missions involve launch of eight astronauts onboard Mars spaceships in December of 2006, in February of 2009, and in January of 2011. Each manned launch is preceded by an appropriate cargo mission. Because they use opposition class trajectory profiles, the manned vehicles spend only 30 days at Mars. On each mission, four crewmembers descend to the surface for up to 20 days of exploration. On the first mission, a Phobos exploration is also performed. On the second mission, both Phobos and Deimos are also explored.

Other requirements and assumptions are given in table 2.2.4-I. Boiloff rates are assumed as given in table 2.1.4-I. The ECCV in this case does not perform a direct entry to Earth, but rather is aerocaptured into Earth orbit, with subsequent recovery of the astronauts and transfer to Space Station Freedom for isolation prior to return to Earth. The same orbital apsidal adjustments described in section 2.1.4.1 are incorporated into the mission profiles of Case Study 2.

Reference System Description.

Configuration and Mass Allocations. Standardized tanks are the same as those in Case Study 1. The interplanetary mission modules (IMM) (habitability package) for this case study are in the "hub-triangle" configuration (figures 2.2.4-1a and -1b), made up of three Space Station Freedom derivative modules with an additional central unit 7.6 m diameter by 3.0 m tall (disk module, 25 ft dia by 10 ft). Three independent entry points are available to each of the four modules. Two separate airlocks are

TABLE 2.2.4-I.- TRANSPORTATION REQUIREMENTS AND ASSUMPTIONS

SRD Requirements:

- Split: sprint/conjunct
- On Mars surf, activate geophys and atmos long-term monitoring exper.
- LEO node used for assy; no nodes beyond LEO (SRD 2.2)
- 1 km landing accuracy (SRD 4.1.1)
- "Minimize the single major sys(s) that could cause to miss a launch period." EOC (SRD 4.1.1)
- EVA's: four 6-hr at Phobos; 10 on Mars; flyby aborts.
- User accom on flight veh: 100 kg, 1 kW (SRD 4.1.1)
- Payloads (A.2): see User Accommodations above

Assumptions for Reference:

- 2-stage TMIS for piloted; 1-stage TMIS for cargo
- Propulsion: Cryo for TMI, TEI, DSM, MOO; biprop for MCC, MOC, RCS; MAV is single-stage biprop.
- Engine performance: Isp = 485 TMI, 470 for other cryo; 320 for storable biprop
- Propellant margins: 1% each for DV, Isp, and bulk (use sum of margins) 3% DV margin on MAV; 2% bulk margin on TEI
- Hab modules: three SS-derived modules plus one disk module ("Hub-Triangle" configuration)
- PVPA for spaceborne power, 300 m^2
- Spaceborne ECLSS: closed for all, except food
- Mars aerobraking; ECCV for crew recovery at Earth Aerobrake technology: very conservative (15%) for piloted; nominal (10%) for cargo
- MOV Mars parking orbit: 250 km x 1 sol; Phobos excursion vehicle (PhEV)
- MDV entry and landing: biprop deorbit, terminal propulsion; aerobraking and parachutes
- MDV habitat: one 7.6 m [25'] diameter disk module
- Landed ECLSS: no O_2, CO_2 recycling; water recycling
- MAV direct to MOV parking orbit (V = 5408 m/s)

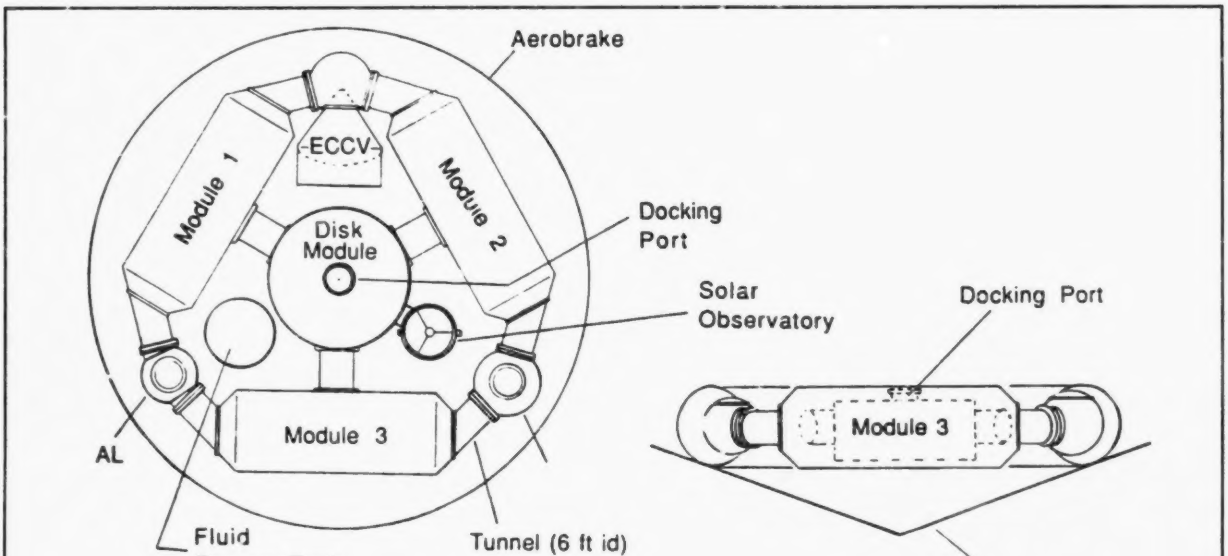

Aerobrake
ECCV
Module 1
Module 2
Disk Module
Docking Port
Solar Observatory
Docking Port
Module 3
Module 3
AL
Fluid Storage Tank Cluster
Tunnel (6 ft id)
Aerobrake

Figure 2.2.4-1a.- Mars transfer vehicle (MTV).

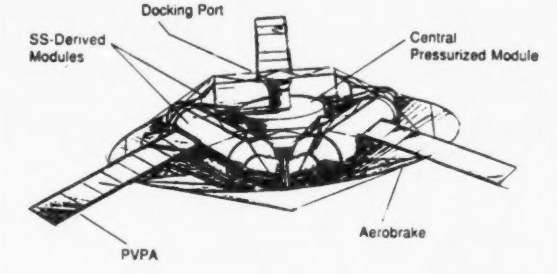


Figure 2.2.4-1b.- Mars transfer vehicle (MTV).

provided, one of which is rated as a hyperbaric lock. A docking port is located on top of the disk module. The ECCV is mounted at one of the intermodule connection tunnels for continuous access. The entire structure is arrayed in a planar configuration and supported by trusswork to the Mars aerobrake (MAb). Two solar flare radiation storm shelters, each accommodating four persons (as described in the **Reference System Description** paragraphs of section 2.1.4.1) are provided, one in module 1 and the other in module 3. A closed-cycle life support system is provided. The IMM, including external services, is 65.9 t. Power is provided by three independent solararrays, providing a total of 19.5 kWe at Mars and higher power levels elsewhere. The MAb is 27.4 m (90 ft) diameter and consists of a hard inner core (33 ft dia) based on Shuttle tile technology, with an outer annulus of flexible thermal insulation (Nextel ceramic cloth). It is expected that the MAb will be initially launched in a folded configuration. The MAb is detailed in table 2.2.4-II.

The cargo vehicle (MCV), figure 2.2.4-2, is similar in appearance to the human-carrier vehicle because of its three cylindrical tanks containing the TEIS cryopropellant. Although the aerobrake needed for this vehicle is much smaller, the same size brake as for the manned vehicle is shown to provide for commonality in design. During the docking and transfer sequence, figure 2.2.4-3, the MCV transfers the TEIS to the piloted vehicle and four crewmembers transfer to the MDV, which nestles inside the TEIS triangle. The MDV is portrayed in figure 2.2.4-4a and -4b. It includes a disk module the same size as that on the MTV (25 ft dia) which serves as the landed habitat. Mars entry and landing are accomplished by aerobraking to a velocity of Mach 2 or less, deploying parachutes, and igniting a bipropellant-based terminal descent propulsion system to provide for a soft landing and up to 1.0 km of crossrange for terminal guidance. Three Delta engines are provided for terminal descent. An unpressurized rover is included for surface exploration. Life support is based on bulk supplies of water and oxygen (stored as hydrogen peroxide) and chemical removal of carbon dioxide. For the 20-day surface mission, a total life support system (LSS) mass of 1.26 t is allocated. The MDV mass is 51.3 t, of which 23.1 t is the MAV, a minimum-mass conical spacecraft which holds four persons, their spacesuits, and 100 kg of returned samples (figure 2.2.4-5). The MAV propulsion system includes 19.4 t of storable bipropellant. Four pressure-fed Delta engines provide the thrust required for lift-off and burn to high elliptical orbit for rendezvous with the MOV.

TABLE 2.2.4-II.-MARS AEROBRAKE DESIGN

Assumptions: C3 = 50 km^2/s^2; P/L mass = 130 t

Aerocapture brake characteristics:		
Diameter		27.4 m[90 ft](Area = 591 m^2 [6361 ft^2])
M/C_dA		143 kg/m^2[29.3 lbm/ft^2]
Angle of attack (alpha)		11.18°
L/D		0.18
Peak deceleration		8.08 g
Mass Summary		
RSI core (33' dia, 0.73" thick)		349 kg
RSI honeycomb substrate		1261
Interface ring		1068
Radial beams		692
Struts		1731
FSI annulus	(90' dia, 1.27" thick)	4590
	Subtotal	9691
	Contingency (20%)	1938
	Total	71,629 kg (8.9%)

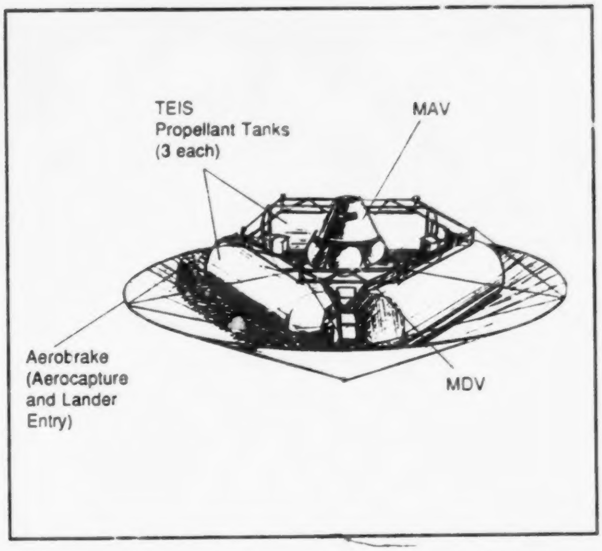

TEIS
Propellant Tanks
(3 each)
MAV
Aerobrake
(Aerocapture
and Lander
Entry)
MDV

Figure 2.2.4-2.- Mars cargo vehicle (MCV).

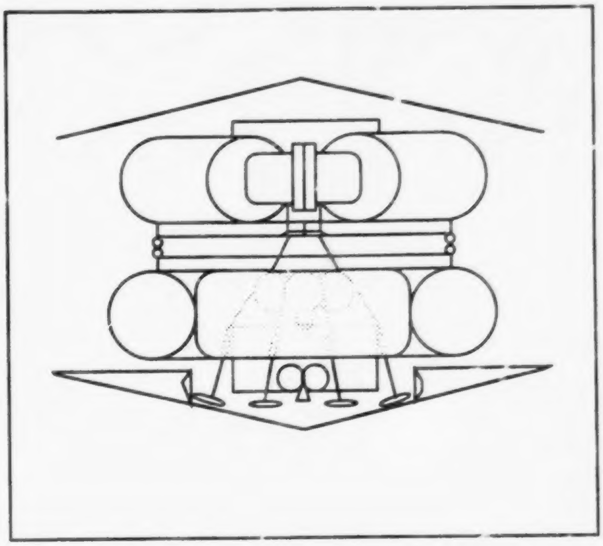


Figure 2.2.4-3.- MTV and MCV docked for transfer in Mars orbit.

Dry Mass	
(includes payload & wet MAV)	47,844 kg
Payload Mass	
(crew & user payload)	13,730 kg
Payload Volume	
MAV	
(cone - 1.8m rad., 2.3m ht)	8 m
HAB Module	
(cyl. - 3.8m rad., 3m ht)	136 m
Propulsion System	
Propellant Type	
Engines	
Number	3
Type	Delta
Mass (ea.)	100 kg
Thrust (total)	134 kN (30.0 klbf)
I (320 sec)	3.14 kN - s/kg
Propellant Mass	3,406 kg
Tank Mass	341 kg
Initial T/W	0.71
Mass Fraction	
Terminal descent	
w/parachute	0.93
Total Mass	51,250 kg

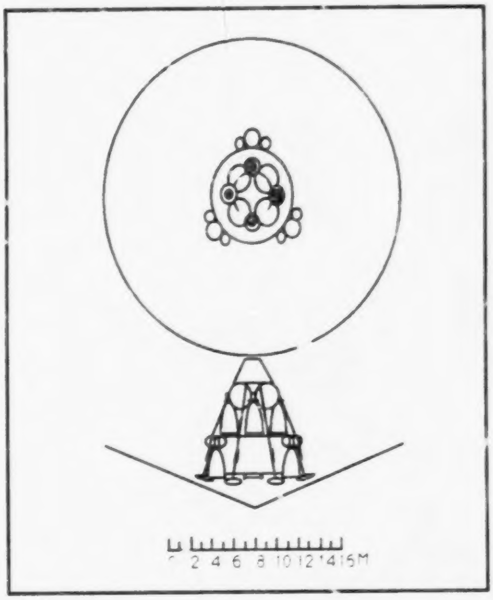

2 4 6 8 10 12 14 16M

Figure 2.2.4-4a.- Mars descent vehicle (MDV).

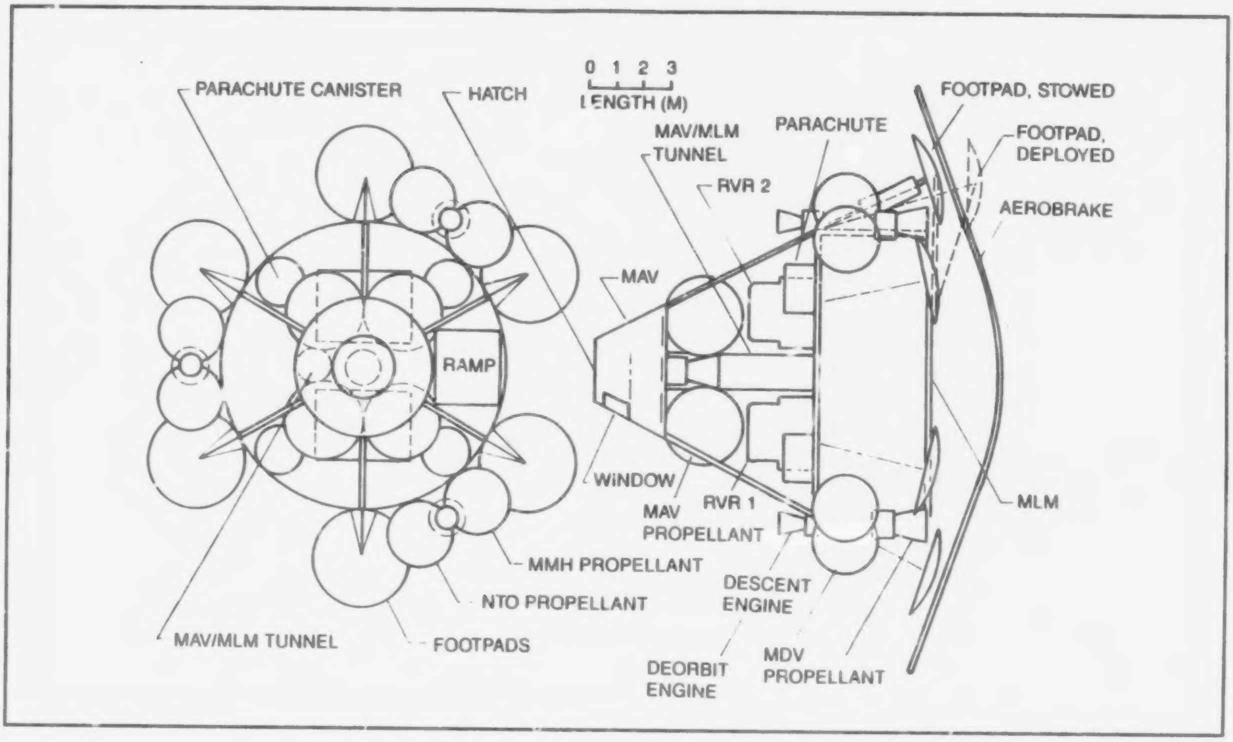

0 1 2 3
LENGTH (M)
PARACHUTE CANISTER
HATCH
RAMP
MAV/MLM TUNNEL
MMH PROPELLANT
NTO PROPELLANT
FOOTPADS
MAV/MLM
TUNNEL
PARACHUTE
RVR 2
MAV
WINDOW
MAV
PROPELLANT
RVR 1
DESCENT
ENGINE
DEORBIT
ENGINE
MDV
PROPELLANT
FOOTPAD, STOWED
FOOTPAD,
DEPLOYED
AEROBRAKE
MLM

Figure 2.2.4-4b.- Mars descent vehicle (MDV).

Dry Mass (includes payload)	3,638 kg
Payload Mass (crew & supplies)	470 kg
Payload Volume	
MAV (cone - 1.8m rad., 2.3m ht.)	8 m^3
Propulsion System	
Propellant Type	MMH/N_2O_4
Engines	
Number	4
Type	Delta
Mass (ea.)	100 kg
Thrust (total)	178 kN (40.0 klbf)
I (320 sec)	3.14 kN - s/kg
Propellant Mass	19,422 kg
Tank Mass	1,942 kg
Initial T/W	2.55
Mass Fraction	
Ascent to 24.66 hr elliptical orbit	0.183
Total Mass	23,060 kg

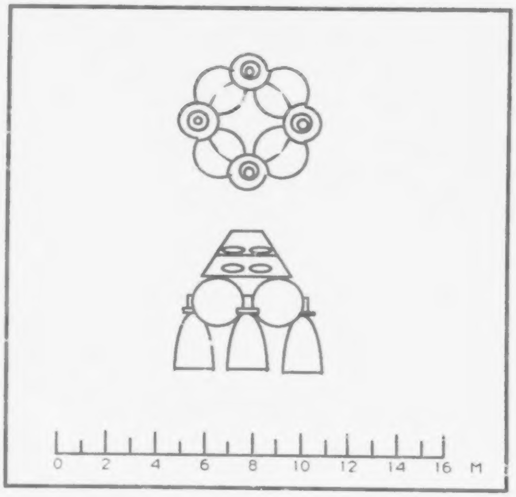

0
2
4
6
8
10
12
14
16
M

Figure 2.2.4-5.- Mars ascent vehicle (MAV).

The TEIS (figure 2.2.4-6) for Mars orbit escape consists of three tanks with a total of 59.5 t of cryopropellant. Six RL-10B-2 engines are provided in a dual triangular array. Only one triad of engines need operate nominally to provide the requisite thrust and thrust-vector alignment during Mars orbital escape. The initial acceleration at Mars departure burn is 0.275 g.

The TMIS for the human vehicle, figure 2.2.4-7, consists of a number of standard tanks with one or more advanced space engines having specific impulse performance of 485 s. Tank number varies with the launch opportunity. MCV payloads include not only the TEIS and MDV, but also satellites totalling 4 t, a 3.5 t MTR, and 0.45 t of onboard solar monitoring and Mars science. The first two missions also carry PhEV's and in the second mission also a DeEV.

The ECCV (figure 2.2.4-8) accommodates eight persons. Its mass is estimated at 9.2 t. It is similar to the Apollo system of a conical capsule and a cylindrical service module portion, but a separate aerobrake is added to facilitate aerocapture. Adequate propulsion to achieve a periapsis raise after aerocapture is provided in the service module. The PhEV is as described in section 2.1.4.1. The DEV is nearly identical except for minor modifications to handle the slightly different propulsion requirements.

Features of the System. The habitability modules and TEI propellant modules are all compatible with Shuttle-C 24-ft diameter payload envelopes. The disk module is compatible with Shuttle-C or ET aft-cargo carrier concepts. Both the MCV and MSS are based upon triangular structures, providing the strongest natural structural design. Hexagonal docking trusses allow double-redundant tripod connection points when the two vehicles dock. The TEIS is a fully contained propulsion system commanded by remote radio link. It is capable of being powered by fuel cells and/or solar array. Engines are fully redundant. Transfer of the TEIS from the MCV to the MSS requires only a mechanical docking; no plumbing or electrical connections need cross this mechanical interface.

The combination of cylindrical and disk modules allows the development of an interior architecture which provides a quality of living appropriate to long duration stays in deep space. Multiple entries enable sealing off any one module without restricting access to the other modules.

ETO, Onorbit Assembly, and Servicing Needs. The number of HLLV launches required depends not only on HLLV lift capability, table 2.2.4-III, but also on the launch year and trajectory selection because of the strong variations in astrodynamic factors from one opportunity to the next (reference table 2.2.1-II). All habitable modules, standard propellant tanks, and the TEIS propulsion units can be accommodated by 25-ft diameter payload shrouds. The system may be assembled as a stand-alone, or at the LEO node. If needed by the assembly crew, early habitation is possible as soon as the aerobrake, the disk module, and one cylindrical module (and its associated photovoltaic power array) are placed into orbit and assembled. In-space propellant transfer is not necessarily required, but because of the large number of launches and the possible stretchouts in assembly time in LEO, top-off propellants from a propellant depot or additional standard tank are desirable. Orbital debris hazards may be mitigated by use of the aerobrakes as forward shields during vehicle buildup.

Transportation Program Development Schedule. The schedule for development, proof-flight testing and man-rating of transportation hardware and propulsion systems is shown in figure 2.2.4-9. As in the previous case study, a number of prior developments are key to the success of this program. It will be critically important to achieve early development of the HLLV, TEIS, and MAV because the capabilities of these transportation systems will affect the derivation of requirements for all other transportation vehicles.

Trades/Options. Several options have been considered and their effects on IMLEO has been calculated. Unless otherwise noted, the trajectories used in the evaluation of these options (and of the system alternatives to be discussed in section 2.2.4.3) were those associated with the 2005 mission opportunity (see table 2.2.1-I), which is no longer considered viable because of development schedule limitations. The total IMLEO for this reference opportunity was found to be 1628 t (see table 2.2.1-II) with the baseline vehicle design parameters. The use of a more realizable TMI engine performance of 480 s (down from 485) causes only a 1.3 percent increase in IMLEO. Backing down the TEI and other non-TMI propulsion performance from 470 to 460 s results in an additional 2.4 percent mass penalty. Cryopropellant storage issues affect IMLEO more profoundly. Use of very conservative tankage factors (dry tank mass/propellant mass) and high boiloff rates would cause an increase in IMLEO of over 50 percent, whereas advanced tankage (7.5 percent factor) can allow an 18 percent reduction in IMLEO. Deleting the manned excursions to Phobos and Deimos, but providing exploration spacecraft teleoperated by the orbiting crew saves less than 1 percent of IMLEO for the three missions. However, manned visits to these moons will significantly complicate mission operations during the relatively short staytimes at Mars and also expose personnel to new hazards without the assurance of any more effective exploration than that accomplished with well-designed robotic freeflyers.

Dry Mass (includes payload)	96,350 kg
Payload Mass (ETV)	86,340 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	6
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)	588 kN
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass	59,480 kg
Initial T/W	1.03
Mass Fraction	
Earth return	0.62
Total Mass	155,830 kg

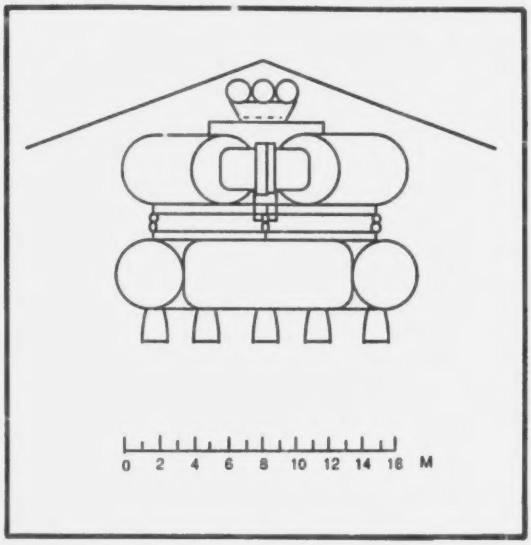


Figure 2.2.4-6.-Trans-Earth injection sysyem (TEIS)-
sprint class mission

Dry Mass (includes payload)	439,140 kg
Payload Mass (MTV)	334,130 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	1
Type	SSME-derivative
Mass (ea.)	3,175 kg
Thrust (total)	2,415 kN
I_{sp} (480 sec)	4.71 kN-s/kg
Propellant Mass	685,330 kg
Tank Mass	21,100 kg
Initial T/W	0.22
Mass Fraction	
Total	0.39
Total Mass	1,124,470 kg

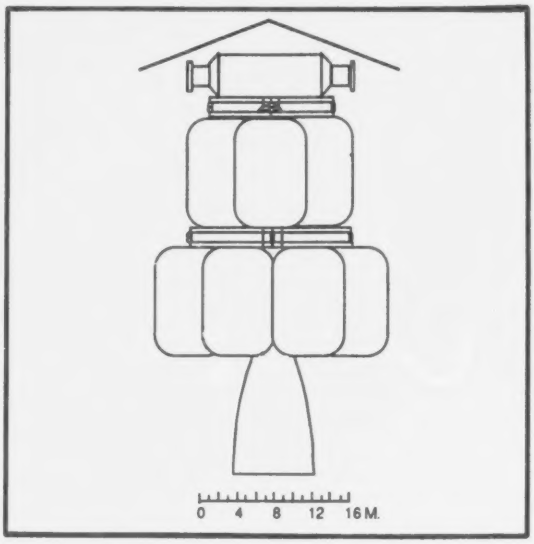


Figure 2.2.4-7.- Trans-Mars injection system (TMIS)-
sprint class mission, piloted (2005).

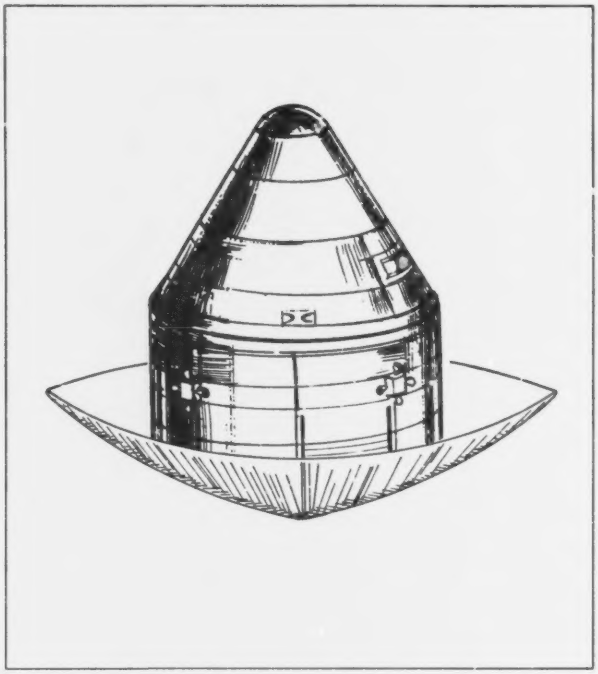


Figure 2.2.4-8.-Earth crew capture vehicle (ECCV)

TABLE 2.2.4-III.- EARTH-TO-ORBIT SEQUENCE*

Year†		'04	'05	'06	'07	'08	'09	'10
Mission	1R	6	12	17				
	2R			5	13	18		
	3R						11	13
Total Launches	6	12	22	13	18		11	13 = 95

* 91 t HLLV; crew launches not included
† Calendar year in which ETO vehicles are launched from Earth to be assembled for subsequent LEO departure.

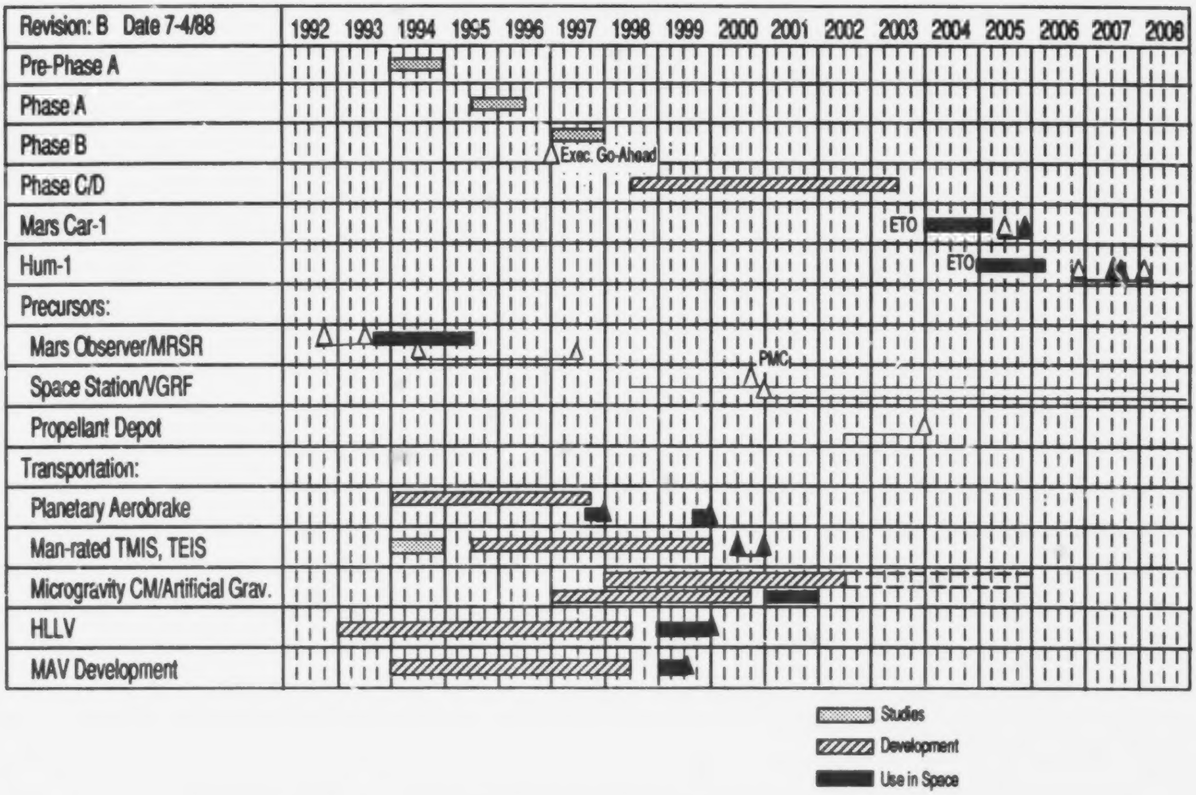


Figure 2.2.4-9.- Transportation program development schedule, Case Study 2.

2.2.4.2 Enabling Technology Needs

As with Case Study 1, both the HLLV and Space Station Freedom (for life sciences research) are required for enabling the transportation systems development. In this case, because of landing on the martian surface, several additional technological developments are needed as well.

Propulsion engines. Development of an advanced cryogenic space engine or space-operated qualification of an SSME-derived engine will be required. Increased performance of the RL-10 engines must be verified (the RL-10B-2 may be an acceptable candidate). Techniques for long-term in-space storage of the RL-10's or equivalent must also be developed and tested.

Cryopropellant tankage. It is quite obvious from the previous discussion that every effort should be made for advancements in cryopropellant storage and for minimizing the tankage mass fraction relative to propellant (the "tankage factor"). This includes consideration of advanced composites, removable structures and shields, efficiency of large multilayer insulation blankets, vapor-cooled shields, and other options.

Mars Descent and Ascent Vehicles. These represent major developments. The MAV, in particular, should be developed and demonstrated very early to provide a solid basis for design of the MDV.

Mars Surface Power. Development of a deployable photovoltaic power array (PVPA) suitable for operation on the martian surface is of high priority because life support and operations for 20 sols will require more power than can be supplied by storage. Issues include broad-range thermal cycling, dust interferences, and methods of deployment. This system will serve as prototype for longer surface stays.

Precursor Missions. Selected missions will be needed to provide spaceborne demonstration and verification of the MDV, MAV, ECCV, IMM, and propulsion systems. Sample return missions to Mars (MRSR's) are a necessity because of concerns for toxicity (biological and/or chemical) and possible reactivity of martian soil. Reactivity could impact lander materials selection, and toxicity could be detrimental to the health of the crew. Current schedules discussed for MRSR do not support the development phase of this case study for launches before 2007.

2.2.4.3 System Alternatives and Opportunities

Elimination of aerobraking for this type of mission results in very large mass increases; for example, it doubles IMLEO for the 2005 reference mission. Utilization of a nuclear thermal rocket (NTR) stage for TMI cuts the IMLEO by one-third. Replacing the cryochemical TEIS stage with a nuclear thermal stage results in only an additional 3 percent gain, however.

Conjunction class trajectories should be given strong consideration for this type of mission because of their capability to lower propulsion requirements as well as increase enormously (up to 25-fold) time available for exploration at Mars. Conjunction class trajectories are sufficiently equivalent that only minor changes in propellant loadings are necessary when launch opportunities are shifted. Also, these trajectories do not travel sunward in their initial stages as do opposition and sprint trajectories, thereby avoiding the higher thermal loads and aggravated solar flare problems (increased radiation levels and reduced warning time). The 2.2 to 2.5 times longer durations of conjunction missions compared to sprints may call for a requirement for artificial gravity.

In figure 2.2.4-10 one concept for a rotating spaceship is shown. Four station-derivative modules are arrayed in the "Bent-I" configuration, connected by tunnels, with a 31-ft disk module at the center of rotation. For the 55-ft swing radius to the floors of the cylindrical modules, the acceleration is up to 0.675 g (achieved at 6 rpm). Decreasing the rate to 4.5 rpm produces the Mars surface gravity of 0.38 g, allowing adaptation by the astronauts to martian conditions prior to arrival at the planet. Even allowing for the larger habitat and the larger diameter aerobrake (135 versus 90 ft), the savings in IMLEO range from 40 to 60 percent, depending on mission opportunity.

Table 2.2.4-IV compares a science-enriched artificial-g/conjunction-class mission with the split/sprint baseline for 2005 mission opportunity. When pressurized habitation volume is increased from 737 up to 1271 m^3, the astronauts live in a more Earth-like environment, more science payload is provided, and the solar cell array no longer has to be deployable/retractable but can be fix-mounted. The mission is "all-up," meaning no rendezvous in Mars orbit is required to obtain the return TEIS propulsion, since it is built into the spaceship. Humans arrive at Mars one and one-half years earlier, with virtually no change in programmatics, except a slightly earlier peak in funding. For a 20 percent increase in IMLEO, but still about 20 percent less than for the sprint case, this vehicle could carry two MDV's. This would allow exploration of two different sites and nearly one year on the surface of Mars, while providing all crewmembers the opportunity to go to the surface and even permitting a rescue of the first landed crew by the second MDV, if it became necessary.

Conclusions and recommendations are summarized in table 2.2.4-V.

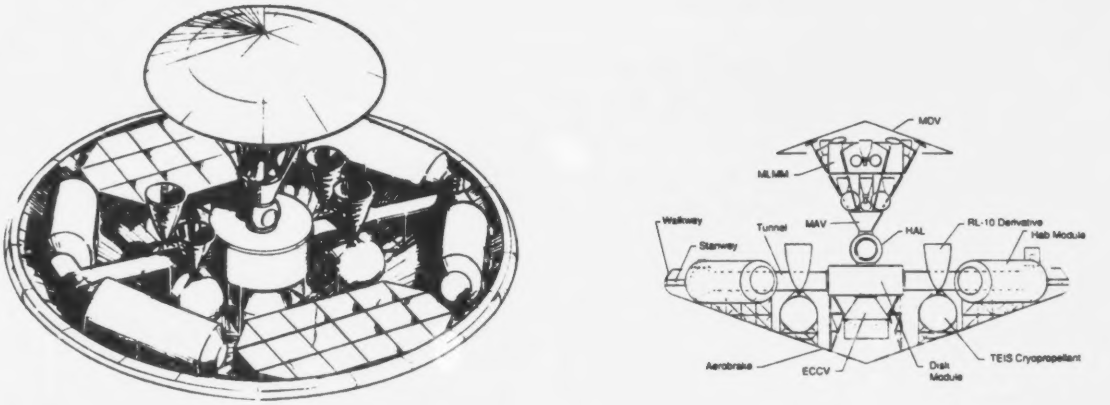


Figure 2.2.4-10.- Artificial gravity Mars spaceship (alternative).

TABLE 2.2.4-IV.- COMPARISON OF ARTIFICIAL g/CONJUNCTION-CLASS AND SPLIT/SPRINT MISSIONS IN THE 2005 REFERENCE OPPORTUNITY

Item	Art-g/Conj [c-HI]	Split/Sprint [c-HG]	Remarks
IMLEO	1090.9 t	1627.8 t	(split: 503.3 cargo, 1124.5 human)
Total trip time	957 d	440 d	
Interplanetary time	204/191 d	265/144 d	
Gravity environment	up to 0.64 g	microgravity	
Arrival date	29 Dec 03	2 Jul 05	
Time at Mars	562 d	30 d	
Time on martian surface	180 sols	20 sols	
TEI propulsion system	integrated	rendez in Mars orbit	
Spaceborne			
No. of SS-derived modules	4	3	
Disk module	31 ft dia.	25 ft dia.	
Other hab space	ECCV MDV MAV tunnels	ECCV	
Total pressurized volume	1271 m^3	737 m^3	conj return leg is 1033 m^3
Power			
Type	fixed	retract/extend	
PVPA area 297 m^2 300 m^2			
Ph/D teleoperators	4000 kg	2000 kg	conj has 2 ea. PhDTele
MarsSciSat	2000 kg	1000 kg	conj has 2 ea. sats
MTR (Rovers/sample return)	two	one	
	8000 kg	4000 kg	
Lander			
Habitat size	31 ft dia.	25 ft dia.	
MLOE			
Science equipment	4000 kg	2600 kg	
Teleoperated equip	2000 kg	2000 kg	
Rovers	two, unpress.	one, unpress.	
mass	2000 kg	1000 kg	
range10 km		10 km	
Construction Equip	2500 kg	250 kg	
Landed power	4.5 kWe	3.5 kWe	
batteries	50 kWh	25 kWh	
Environment			
Dust storm season?	no (Ls=50°)	possible (Ls=325°)	
Recovery of spaceship (at Earth)	yes	no	
Mass penalty for 2nd MDV	232.8 t	not practical (for 64.9 t MDV)	

TABLE 2.2.4-V.- TRANSPORTATION CONCLUSIONS/RECOMMENDATIONS

Programmatics

Parallel development of
- Spaceship (hab modules, external services)
- SS, HLLV, microgravity countermeasures
- Earth-orbit retrieval of ECCV

Flight demonstrations of
- Planetary aerocapture
- Man-rated TMIS/TEIS
- Mars descent and landing (esp. parachute assist)

Enabling/enhancing technologies

Enabling:
- Designs which are compatible with onorbit assembly technology
- Zero-gravity countermeasures effectiveness
- Mars aerobrake for aerocapture
- NTR propulsion (option)

Significantly enhancing:
- Advanced, lightweight, and low-boiloff tanks; dual wet-dry launch (Siamese twins)
- Lightweight aerobraking (reusable brakes; ablators)

Interfacing

LEO node servicing (OMV; construction crew; assembly)
In-flight communications (Earth-to-Mars)
Surface operations and requirements
Connection/requirements with overall transportation infrastructure

Issues

Reliability
- HLLV
- Duplicate hardware builds (flight units #1, #2, etc.?)
- Abort modes

Aerobrake technology timelines
NTR technology timelines (option)
Mission contingencies
- Human safety
- Mission success

Precursors

Transportation man-rating
Technology demonstrations in space (Ab, ECLSS, etc.)
Mars missions, including sample return and landed navaids

Challenges of split missions

Rendezvous in Mars orbit
Propellant transfer

Lessons learned

IMLEO sensitivity to split missions vs conjunction class missions
Operations penalties and dangers of Phobos and/or Deimos missions (combined with landed missions)

2.2.5 Orbital Node Systems Definition

The Human Expeditions to Mars Case Study involves six vehicle flights (three cargo and three piloted) to Mars over three consecutive opportunities beginning in 2005. The LEO transportation node serves as the assembly and staging base for all vehicles. Because the vehicles are massive (dry mass of the piloted vehicle is greater than 150 t), they must be assembled at the transportation node. This involves mechanical, electrical, and fluid connections as well as construction or assembly of vehicles and elements (such as large aeroshells). The cargo vehicle for a given opportunity is launched approximately 19 months before the piloted vehicle to ensure that it has reached Mars orbit and all systems are operational before the piloted vehicle launch from Earth orbit. The mission spacing thus allows assembly and checkout of only one vehicle at a time at the transportation node.

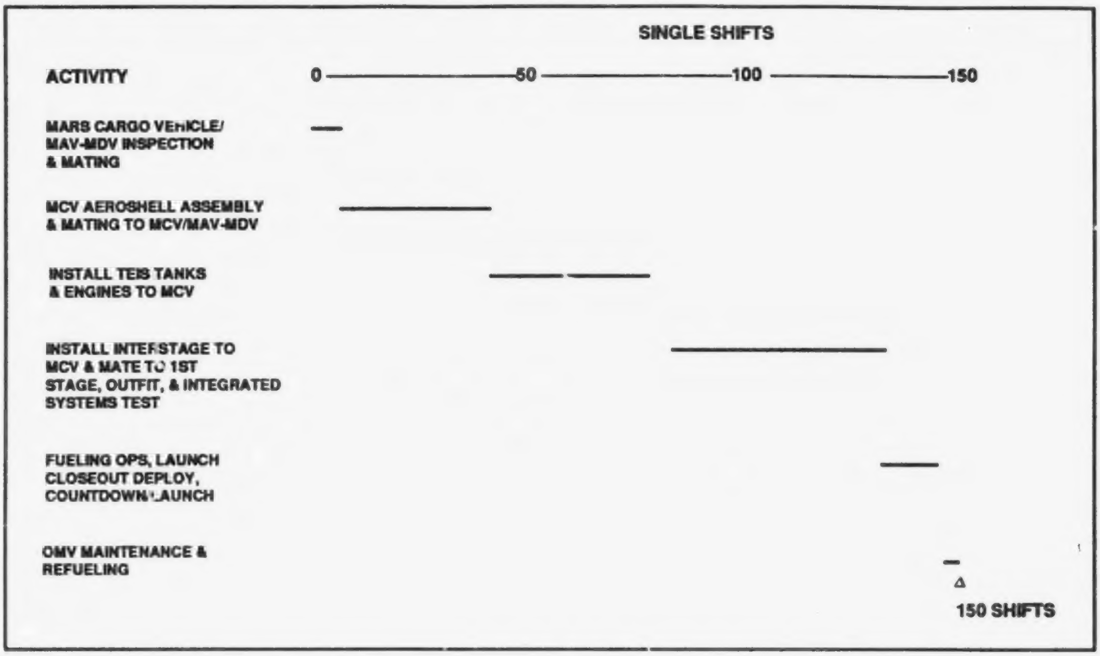
2.2.5.1 Elements and Systems Description

The location of the transportation node is influenced by a number of factors. A primary concern, maximizing mass from Earth to orbit, implies a due east launch from Cape Kennedy to a low altitude orbit. Because atmospheric drag results in orbit decay, it is necessary to have the orbit altitude sufficiently high to keep reboost requirements within reasonable bounds. These considerations lead to an orbit inclination of 28.5° and an altitude in the 150 to 250 nautical mile range.

To understand the onorbit assembly and checkout process, actual launch vehicle processes at the Kennedy Space Center were reviewed. Activities that must occur at the transportation node were then established. Because space vehicle processing on the ground is tracked in terms of days or shifts, that policy is continued in this analysis. Clearly one cannot move the current processing function from the ground to orbit. The cost of maintaining a several hundred- to a thousand-person crew onorbit is obviously prohibitive. To reduce crew time, major advances in onboard automation must be achieved, particularly for self-checking and fault-tolerant systems. Substantial advances in telerobotics, including end effectors with force feedback, are also required. With system and vehicle designs which focus on minimal onorbit processing, the timelines given in tables 2.2.5-I and 2.2.5-II are likely to be achievable. A shift is defined as a crew of six working for 9 hours. It is assumed that two crewmen are at the command center and four are either EVA or working through telerobotics.

Processing the cargo vehicle requires 150 shifts to carry out the functions noted in table 2.2.5-I. The piloted vehicle, with all the crew systems onboard, requires substantially greater processing, totaling 225 shifts. It is

TABLE 2.2.5-I.- CASE STUDY 2, MARS CARGO VEHICLE (MCV) ONORBIT PROCESSING



assumed that the activities listed in tables 2.2.5-I and 2.2.5-II will be conducted onorbit. More detailed analyses are currently underway to establish specific resources requirements and automation and robotics technology/ systems requirements.

The transportation node functions can be accommodated at the Space Station Freedom in a desirable orbit, or a separate facility could be established in LEO. A number of intermediate options, noted in table 2.2.5-III, range from full accommodation at Freedom Station (option 1) to a completely branched facility (options 4 and 5) which has no interaction with the baseline Freedom Station. An intermediate option is assembly at the station with propellant storage and handling at a coorbiting propellant tank farm (option 2). Option 3 uses a coorbiting platform as the assembly and propellant storage base, with the crew quarters at Freedom, and requires a space-based and man-rated OMV with a crew cab.

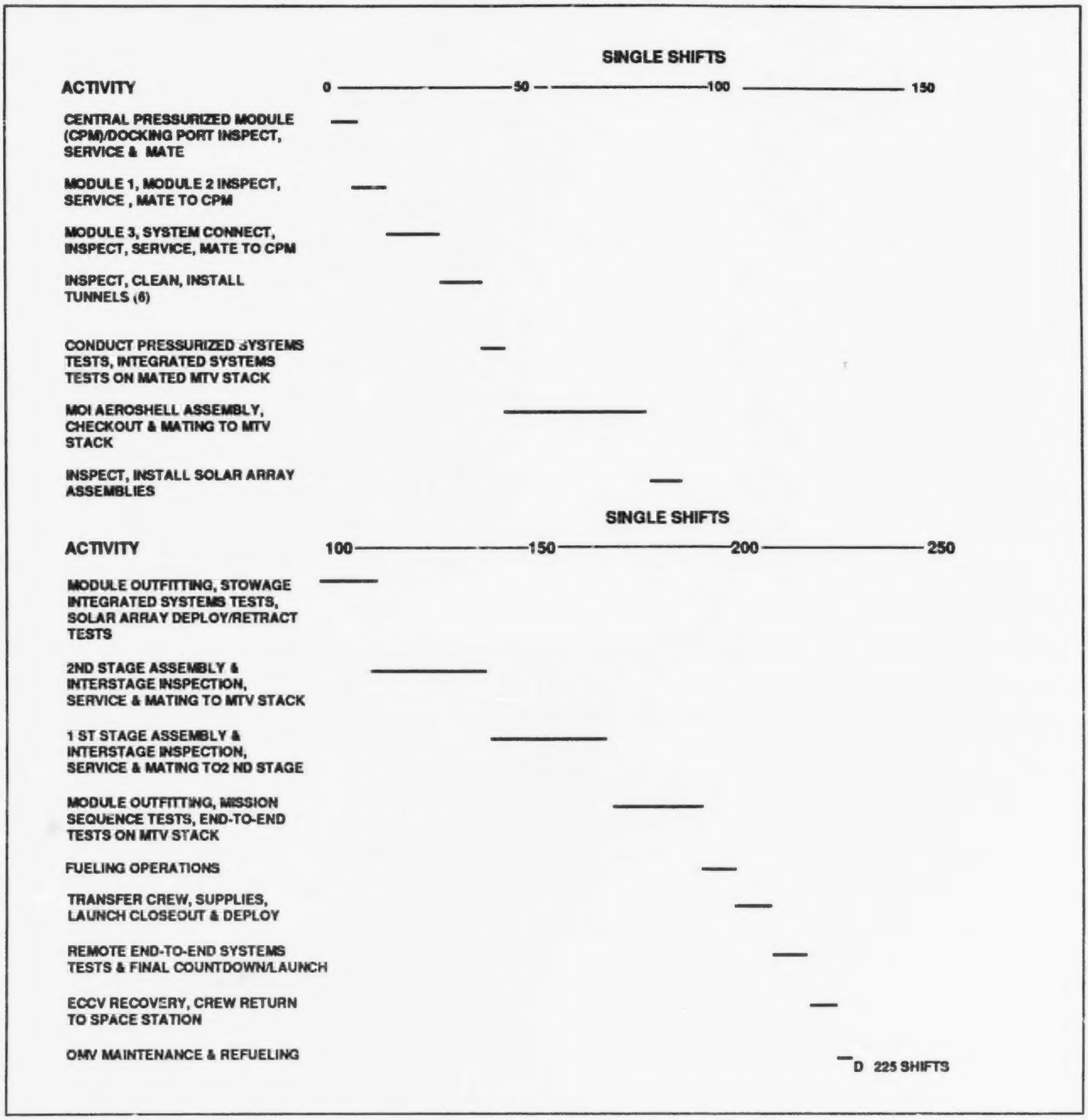
Prior to establishing transportation node accommodation of the Mars mission, the following ground rules and assumptions were established.

a. Phase 1 Space Station Freedom configuration is used as the baseline.
b. Life sciences research will be conducted on the Space Station Freedom.
c. A HLLV with a 91 t payload lift capability is available.
d. A space transfer vehicle capable of delivering/maneuvering 91 t in LEO is required.
e. Mars mission vehicles are assembled and verified in LEO, one vehicle stack at a time.
f. Two man-rated OMV's are available for routine crew transfer if remote assembly facilities are necessary (vehicle accommodation option 3).
g. Liquid oxygen and hydrogen are the propellants for the Mars vehicles, hydrazine for the OMV's (with cold gas jets for Space Station Freedom proximity operations).

Design features requisite to establishing transportation node capabilities and concepts for accommodation of the Mars mission vehicles include:

a. Size and volume to accommodate Mars vehicles and support equipment.
b. Pressurized "command center" for controlling and monitoring EVA and robotic activities.
c. Capability for expansion.
d. Robotic and EVA access to vehicle and propellant tanks.
e. Simple vehicle egress/separation.
   (1) Vehicle egress along velocity vector or negative radius vector.
   (2) Room to avoid collisions with structure.
f. Micrometeoroid/impact protection for vehicle, EVA crew, and propellant by maximum vehicle enclosure.

TABLE 2.2.5-II.- CASE STUDY 2, MARS TRANSFER VEHICLE (MTV) ONORBIT PROCESSING



g. Enclosed volume to contain debris produced by vehicle processing operations.

h. Thermal protection for EVA crew and propellant.
   (1) Volume enclosure if possible.
   (2) Propellant tank shielding.

i. Depot-specific features (accommodation options 4 and 5).
   (1) Docking facilities to accommodate OMV and Shuttle.
   (2) Room for propellant tanks (except option 5) and support equipment.
   (3) Solar dynamic power system.
   (4) Guidance, navigation & control (GN&C), communications & tracking (C&T), reaction control system (RCS) systems.

In addition to the design features listed above, performance goals were established for all concepts. These include:

a. Controllability of all phases of vehicle assembly.
   (1) Minimum control system size and complexity.
   (2) Minimum torque equilibrium attitude (TEA) to maintain protection envelope, ease of separation, and viewing angles.

b. Orbit decay of station/depot and vehicle.
   (1) Decay rates permitting safe separation.
   (2) Minimum reboost propellant needs for all phases of vehicle assembly.

Time was insufficient in this initial study for detailed analysis of all the options listed in table 2.2.5-III. Two of the more promising options (2 and 3) were selected for deeper analysis, which is summarized in the following paragraphs.

Accommodating the precursor research program and the assembly/checkout functions requires a doubling of Space Station Freedom mass (for accommodation option 2) as indicated in table 2.2.5-IV for two concepts. In addition, the Mars vehicles themselves, without propellant, are both large and massive, bringing the total mass at the station to a level of 900 t. The mass summary for a third concept, based on accommodation option 3, is given in table 2.2.5-V. The open box configuration appears to be somewhat less massive than the additions to the baseline Space Station Freedom given in table 2.2.5-IV. It must be noted, however, that the additional life sciences and technology development activities, as well as the two additional hab modules and associated nodes, must still be added to the baseline station. The servicing facility and OMV's are also required at Space Station Freedom to transport crew and HLLV payloads. Thus the total mass in space to support the Mars mission is substantially greater (approximately 20 percent) for the accommodation option 3. Open box configurations are shown in figures 2.2.5-1 through 2.2.5-3.

2.2.5.2 Technology Drivers

For the LEO node systems definition, the required technologies, both enabling and enhancing, are those that provide the capability to assemble, process, and service the particular mission vehicle(s) in space. In this analysis, the node support requirements were divided into the categories of onorbit assembly and onorbit vehicle processing so that specific onorbit resources such as crewtime, power, utilities, facilities and supporting infrastructure could be quantified. Depending on the number and type of flights/reuse inherent in the case study mission design, this categorization allowed the in-space tasks to be classified further as either recurring or nonrecurring to aid in defining the LEO node and supporting systems requirements when additional indepth studies are initiated. Cryogenic fluid management and autonomous rendezvous and docking are also discussed herein, but only from the standpoint of their effect on the onorbit operations.

TABLE 2.2.5-III.- VEHICLE ACCOMMODATION OPTIONS

Option 1:	Space Station Freedom based	- All vehicle accommodations based on station
Option 2:	Space Station Freedom based w/PTF	- Vehicle assembly and refurbish ment facility is on station. Propellant is located on a coorbiting propellant tank farm (PTF)
Option 3:	Transportation depot (man-tended)	- Vehicle accommodations are kept on a coorbiting platform, but crew is based on Space Station Freedom
Option 4:	Transportation depot (permanently manned)	- A separate facility is provided for vehicle and crew
Option 5:	Transportation depot	- Vehicle assembly accommodations and crew facilities are provided off Space Station Freedom. In addition, a coorbiting PTF is used to store and reliquefy propellant onorbit.

TABLE 2.2.5-IV.- SPACE STATION FREEDOM GROWTH HARDWARE TO ACCOMMODATE HUMAN EXPEDITIONS TO MARS

COMPONENT	CONCEPT A NUM.	CONCEPT A MASS (kg)	CONCEPT B NUM.	CONCEPT B MASS (kg)
Truss bays	101	7,400	106	7,800
Utility bays	101	22,700	106	23,600
Solar dynamics	6	33,100	6	33,100
Habitat module	2	39,000	2	39,000
Node	4	23,300	6	34,000
Laboratory module	2	62,900	2	62,900
Pocket labs	2	10,200	2	10,200
Servicing facility + 2 OMV's (wet)	1	22,400	1	22,400
Standard airlock	1	2,000	1	2,000
Hangar + Equipment	1	25,000	1	25,000
Total additional hardware		248,000		260,000
+Phase 1 station 210,000 kg		48,000		470,000
+Mars cargo vehicle 231,000 kg	1	689,000	1	701,000
or +Mars piloted vehicle 439,100 kg	1	897,100	1	909,100

TABLE 2.2.5-V.- BRANCHED TRANSPORTATION NODE MASS SUMMARY — OPEN BOX CONFIGURATION

Component Name	Mass (Kg)
Airlock	2,014
Alpha Joints	1,200
CMG's	1,567
Cupola	1,455
Docking Adapters	1,000
Nodes (2)	9,091
Command Center	31,523
MSC/Transporter	4,909
RCS Clusters	1,025
RCS Propellant & Tanks	16,364
Solar Dynamic Power Modules (2)	14,078
TDRSS & Antenna	586
Teleoperated Servicer (2)	2,381
Propellant Storage Tanks (11)	68,924
Attached Hardware	12,980
Radiators (2)	3,67-
Logistics	8,285
Truss	9,875
Utility Trays	18,008
Total	198,900

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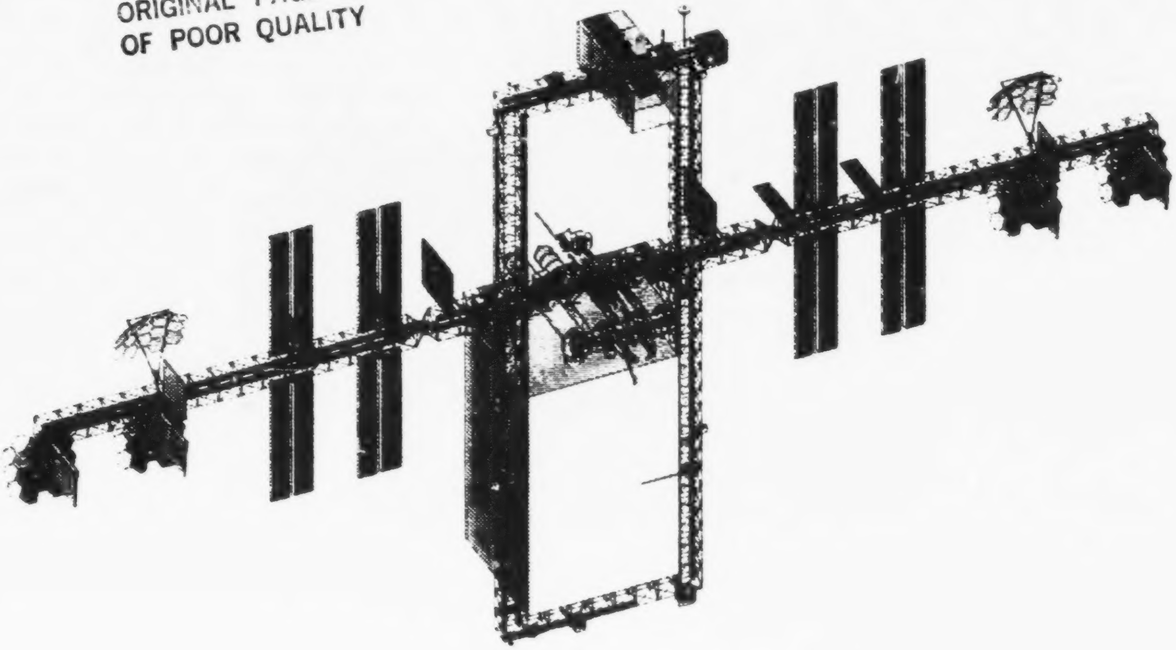

Figure 2.2.5-1.- Concept A, Mars mission accommodation.

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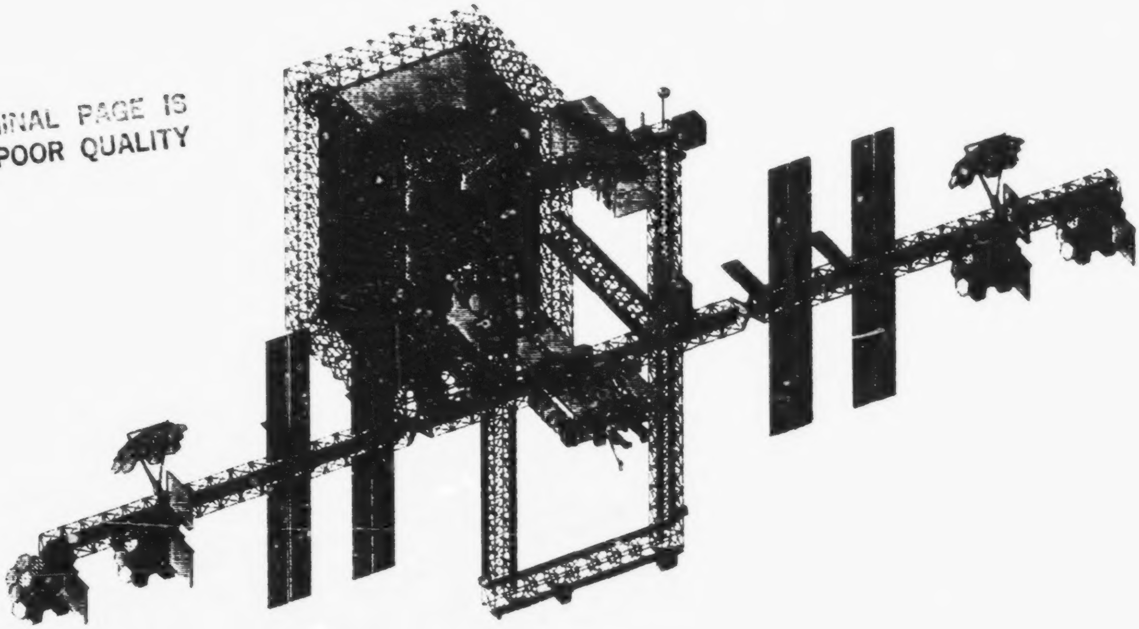


Figure 2.2.5-2.- Concept B, Mars mission accommodation.

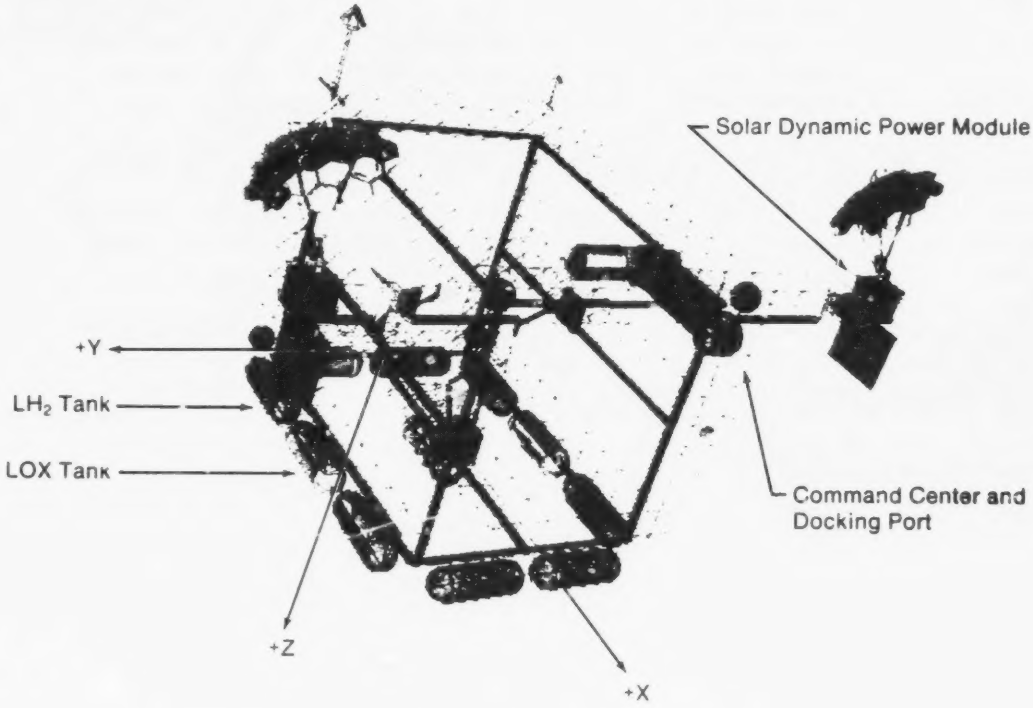

Solar Dynamic Power Module
+Y
LH_2 Tank
LOX Tank
Command Center and Docking Port
+Z
+X

Figure 2.2.5-3.- Man-tended transportation depot (open box concept).

A commitment to provide an extensive LEO node onorbit assembly and vehicle processing capability will require considerable future study effort. However, two important factors will undoubtedly influence the decision to provide this capability: the specific mission designs and the performance characteristics of ETO launch systems.

To accommodate onorbit assembly at the LEO node, the capability to assemble, handle and mate/demate very large, very heavy, and complex space vehicles will be required. A high degree of confidence and reliability must be demonstrated, and assembly operations must be conducted with minimum risks and minimum intravehicular activity (IVA)/EVA crew involvement. For the planetary space vehicles, space propulsion systems, and any reusable elements/injection stages, the onorbit technology program must address handling, assembly, and mating techniques using large capacity, highly articulated manipulators and telerobotic/teleoperated aids. The success of providing this capability depends on major technological advances in the areas of automation and robotics, autonomous rendezvous and docking, and control of large structures.

In addition, spacecraft aerobraking systems (e.g., large diameter aeroshells) assembly operations will require special handling and assembly techniques due to the close tolerance requirements and possible fragility concerns inherent in aeroshell thermal protection systems (TPS). The onorbit technology program required to support the aerobraking systems must address the handling and assembly techniques associated with the joining of structural elements by such means as welding, bonding, and snap-connectors; the attachment/application of advanced TPS components/insulation; and the removal and refurbishment of TPS materials and structural elements.

The onorbit vehicle processing function, while requiring many of the attributes needed by the orbital assembly function (i.e., handling, mating, and manipulating large and massive mission elements in space), must also be capable of integrating, testing, and subsequent checkout of any and all elements of the space vehicle. To accomplish onorbit what has always been done with ground-based facilities will require a whole new set of operational philosophies, procedures, and support equipment, especially where manned systems are involved. Trade studies have been initiated to address these issues. However, from a technology standpoint, the orbital test programs for this function must focus on the development and implementation of advanced systems capable of performing automated checkout and systems status interrogations on each element as processed, and on the final flight configuration.

From the orbital node viewpoint, the capability to handle, transfer, and manage large quantities of cryogenic propellants in space for long periods of time must be developed and demonstrated onorbit before these missions can seriously be considered. The facilities and techniques required to transfer the propellants from tank to tank and tank to vehicle with minimum boiloff and contamination in and around the LEO node and mission vehicles must be available early in the programs to be incorporated into the LEO node system definition and design.

Autonomous rendezvous and docking is another key technology driver in implementing the proposed case study missions. Space-based systems must be developed that are capable of autonomous rendezvous and docking with very large, very heavy and massive vehicles such as ELV's, mission vehicles, and reusable transfer vehicles and injection stages. The system must be capable of stabilizing and maintaining control of these mission elements for subsequent handoff and transfer to the station, node, and/or coorbiting facilities with a high degree of accuracy. This capability must further be incorporated into an OMV-type system specifically tailored to handle large masses, (>100 t), with adequate control authority to deliver and retrieve mission elements to and from staging orbits.

2.2.5.3 System Alternatives

Our studies have focused on a zero-g transportation system for the Mars mission. If life sciences research is not able to develop adequate countermeasures, an artificial-g transportation system will be required. A concept of such a vehicle is described in section 2.2.4.3. While this system is slightly more massive than the zero-g configuration, it does not appear to pose any additional major problems for the transportation node function. This option will be examined in greater detail in FY 1989.

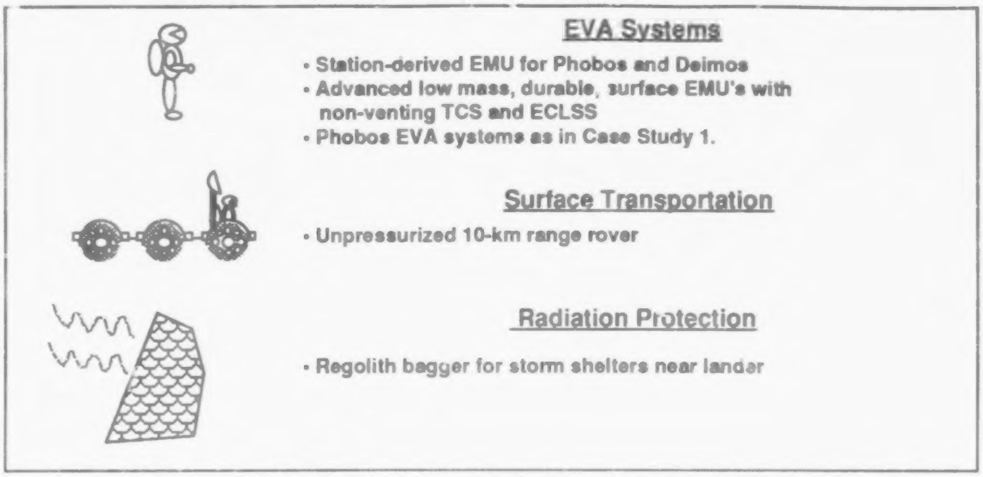
2.2.6 Planetary Surface Systems Description

Case Study 2 deploys science packages on the surfaces of Mars and Phobos in a manner similar to the Apollo missions. The primary purpose of the surface elements in these expeditionary missions to the Mars system is to support the crew while they explore the surfaces.

2.2.6.1 Elements and Systems Description

Figure 2.2.6-1 summarizes the surface elements identified to support Case Study 2. In addition to those identified in the previous section for the exploration of the surface of Phobos, the list includes three new systems: the Mars surface EVA systems, an unpressurized rover, and a regolith bagger to provide emergency radiation protection.

The Mars surface EVA systems envisioned in these studies are extrapolated from current EVA information. The suits are extremely heavy with a mass of 230 kg, equivalent to 190 pounds on Mars. In addition, the suits are



EVA Systems

- Station-derived EMU for Phobos and Deimos
- Advanced low mass, durable, surface EMU's with non-venting TCS and ECLSS
- Phobos EVA systems as in Case Study 1.

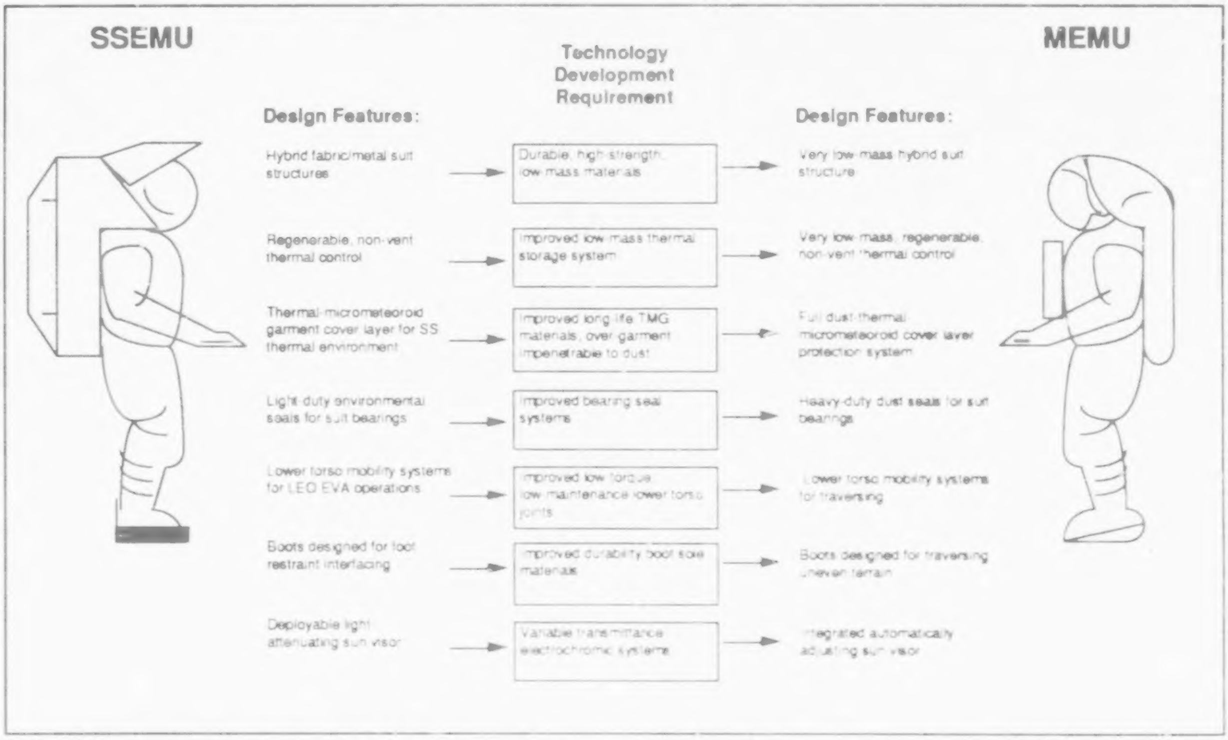
Surface Transportation

- Unpressurized 10-km range rover

Radiation Protection

- Regolith bagger for storm shelters near lander

Figure 2.2.6-1.- Planetary surface systems elements for Mars expeditions.



SSEMU

MEMU

Design Features:	Technology Development Requirement	Design Features:
Hybrid fabric/metal suit structures	Durable, high-strength, low mass materials	Very low-mass hybrid suit structure
Regenerable, non-vent thermal control	Improved low-mass thermal storage system	Very low-mass, regenerable, non-vent thermal control
Thermal-micrometeoroid garment cover layer for SS thermal environment	Improved long life TMG materials, over-garment impenetrable to dust	Full dust-thermal micrometeoroid cover layer protection system
Light-duty environmental seals for suit bearings	Improved bearing seal systems	Heavy-duty dust seals for suit bearings
Lower torso mobility systems for LEO EVA operations	Improved low torque, low maintenance lower torso joints	Lower torso mobility systems for traversing
Boots designed for foot restraint interfacing	Improved durability boot sole materials	Boots designed for traversing uneven terrain
Deployable light attenuating sun visor	Variable transmittance electrochromic systems	Integrated automatically adjusting sun visor

Figure 2.2.6-2.- Mars EMU technology development requirements.

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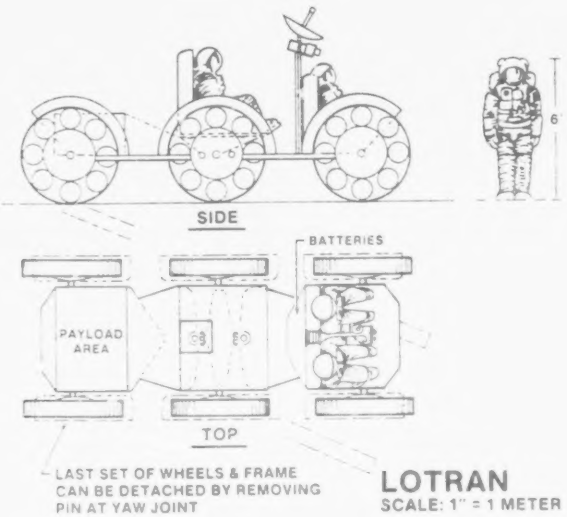
6'
SIDE
BATTERIES
PAYLOAD AREA
TOP
LAST SET OF WHEELS & FRAME CAN BE DETACHED BY REMOVING PIN AT YAW JOINT
LOTRAN
SCALE: 1" = 1 METER

Figure 2.2.6-3.- Mars unpressurized rover.

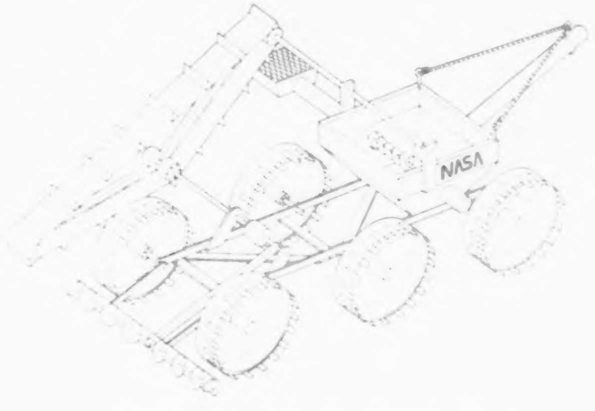

Figure 2.2.6-4.- Mars regolith bagger.

assumed to be more flexible than current or Apollo suits and are assumed to be capable of supporting daily EVA's. Some additional features of a Mars EVA system versus a Space Station Freedom EVA system are illustrated in figure 2.2.6-2.

The unpressurized rover must have the capability to traverse out 10 km from the base and return. The concept for the rover is shown in figure 2.2.6-3. It is equipped to carry two crewmembers but can be outfitted to handle four, and it has the capability to carry 570 kg of equipment. The rover is powered by 4.3 kWe supplied by rechargeable batteries and is configured such that all wheels are motor driven.

A regolith bagger, as shown in figure 2.2.6-4, will be brought along to provide radiation protection around the base of the lander during solar flares. Shielding thicknesses of 1 m have been estimated to provide Earthlike radiation protection from solar events. Not estimated in these calculations was a means of achieving similar protection for a crew aboard the rover that was more than one-half hour from the lander. In such cases, the crew would not have sufficient time to return to the base before the radiation level would begin to increase from the solar event. As a matter of course in the mission, the crew would use the bagger and equipment to put the shielding around the bottom of the lander to make the radiation-safe haven. Overall, the use of regolith on Mars is an excellent option due to the additional shielding provided by the local atmosphere.

The main purpose of these surface systems is the support of the science mission. For the purposes of this study, the following Mars surface science capabilities were assumed.

a. Teleoperated rover
b. Meteorological balloon
c. Biology experiment package
d. Geophysical/meteorological station
e. Sample collection sets
f. Subsurface core drill (attached to rover)

The result of this set plus the human and rover abilities provide the ability to explore Mars thoroughly within 10 km of the landing site.

2.2.6.2 Technology Drivers

There is one major technology need in this case study: EVA systems and techniques. Other issues are the production of power for the rovers and the need to ensure adequate warning of solar events to provide sufficient radiation protection.

The major problem with the current design for the Mars surface EVA system is the mass. The human occupant will be restricted in motion and will also have a strenuous time moving due to the extra inertia of the suit. New materials and techniques for such systems must be designed. Also, these suits should be highly resistant to a dusty environment and should provide easy cleaning to prevent internal contamination upon return from an EVA into the habitat area. An idea that deserves further study is the possibility of mobility assistance devices on the surface.

With the current design, rechargeable batteries or fuel cells are adequate for the 10-km traverses. However, with improved technology, that is, more power for less weight, the ability of the rover can be increased. Also, the ability to provide lightweight portable radiation protection should also be studied on Mars to allow the crew to get farther from the rover and still be able to withstand a solar event.

2.2.6.3 Systems Alternatives and Opportunities

Alternative schemes for radiation protection and the use of local material for surface vehicle power have been suggested and deserve further study for expedition-class missions to Mars.

Radiation protection garments developed from multilayered carbon fibers can be worn over suits to reduce exposure to potentially lethal doses of radiation while allowing the crew member to move about. Current designs reduce the dose rate from a solar flare by five to seven times. These garments would not usually support the crew throughout an entire flare but would give added time for more appropriate measures.

Carbon monoxide and oxygen extracted from the martian atmosphere could be used in a zirconia-based fuel cell that would provide power for surface vehicles. This same cell can be used to extract oxygen from atmospheric carbon dioxide to provide propellant or life support.

2.2.7 Case Study Synthesis

2.2.7.1 Evaluation of Inputs

Consistency. There is a slight inconsistency associated with the MDV payload capacity. The Transportation IA designed the MDV for 6.57 t of support and science payload.

Mars landed science equipment	2.30 t
Mars landed transportation equipment (includes rovers and suits)	1.02 t
Teleoperated equipment	3.00 t
Mars landed construction equipment	0.15 t
Mars landed manufacturing equipment (includes in situ resource utilization demo)	0.10 t

The Planetary Surface Systems (PSS) IA planned for 10.96 t of support and science payload (table 2.2.2-I). Thus, the MDV is supposed to carry 4.40 t more payload than it was designed for. The manifest in table 2.2.2-I uses the PSS payload mass numbers; figure 2.2.4-4b assumes the TA numbers. This inconsistency will not alter the basic design characteristics of the MDV and is well within the margin of error expected for any space transportation vehicle point design at this early stage of analysis.

Parametric Results. For the most part, data were not submitted in parametric form. Although sufficient detail was provided in some areas for parametric relationships to be deduced, more explicit information is needed to support FY 1989 studies.

Options. The options explored by the Transportation IA in attempting to meet the LEO mass limitations were appropriate, and yielded useful data for the guidance of future studies.

Special Assessments and Broad Trades. Many of the special assessments and broad trade results have only limited application to this case study because of its requirement to minimize reliance on new technology. Two exceptions are the special assessment of solid core nuclear thermal rockets (NTR's) and the broad trade study of end-to-end interplanetary spacecraft assembly in LEO. The latter was not completed in time to be useful to the Transportation IA, but the former was put to good use in exploring the option of using nuclear thermal in lieu of chemical rockets, thereby reducing the LEO departure mass of the Mars expedition spacecraft.

2.2.7.2 Principal Issues and Program Risks

The remarks contained in section 2.1.7.2, under the following paragraph headings, are believed to apply to this case study also: Zero-g Countermeasures, Mars Aerocapture vs. Nuclear Thermal Rockets, ETO Payload Capacity of HLLV.

Deep-Space Maneuvers. The sprint trajectories selected for the crew carriers in this study routinely make use of deep-space propulsive maneuvers (DSM's) of considerable magnitude to shape the outbound heliocentric flightpath. This is an effective countermeasure against the effects of Mars orbit eccentricity on opposition-class mission delta V requirements, and a virtual necessity for most sprint (440-day) missions. However, if the DSM, like TEI, must be executed successfully before the flightcrew can return to Earth (which is presumed to be the case), incorporating it into the profile adds a significant amount of risk to the mission.

Sprint vs. Conjunction-Class Mission Profile. Even though DSM's reduce the variation of sprint delta V requirements from one mission opportunity to another, the residual variation is still very significant and introduces much complexity into spacecraft design and operational planning as well. Given also the fact that the available time at Mars is quite short for any reasonable opposition-class trajectory (of which the sprint flighpaths are a subset), many analysts over the last 25 years have come to the conclusion that conjunction-class trajectories — even though they double the round-trip time — have to be the basis of any really effective plan for early human exploration of Mars. The data generated by the Transportation IA with regard to a conjunction-class option in this case study provide valuable information for planning FY 1989 study activities.

2.3 LUNAR OBSERVATORY (CASE STUDY 3)

Case Study 3 emphasizes the long-term acquisition of lunar surface, lunar environment, astrophysics, and astronomy data toward a quantum advance in our knowledge of the Moon, the solar system, and the universe.

The lunar surface offers an environment that makes it an attractive place to put major astronomical facilities. It provides a stable, slowly rotating high-vacuum environment whose far side is permanently shielded from Earth. The scientific expectations for this class of research facility are numerous. For example, a very low frequency radio array affords the capability to map long wavelength radio sources, observe nonthermal radiation from plasma instabilities, and study previously unobserved features associated with galactic sources. A very large optical interferometer array could provide stellar analogs of the solar cycle and sunspots from high resolution star images, characterize the planets of nearby stars, and improve understanding of quasars, galactic nuclei, and black holes through increased resolution.

2.3.1 Case Study Overview

2.3.1.1 Key Features

The principal feature of this case study is the emplacement and operation of a group of astrophysical observational instruments. The observatory will be located on the far side of the Moon to avoid radio interference from the Earth.

In addition to the observatory on the far side of the Moon, manned missions to other sites will provide detailed geological and geophysical data for locations of interest on both the near and far sides of the Moon.

The overall complexity of the surface facility systems and scientific instrumentation is reduced through appropriate use of humans for emplacement, operations support, and periodic maintenance and servicing.

There is a precursor requirement for the acquisition of global lunar environmental, topographical, cartographical, and geophysical data for observatory site selection and for the periodic sortie missions to other sites.

No additional human space physiology research is required for this case study because the transit and surface stay times are short and Apollo program data are available and applicable.

2.3.1.2 Mission Profile

Setup of the observatory will be accomplished over a 2-year period beginning in the year 2004. A split approach is used during the setup period. One unmanned cargo mission is flown in 2004 and another in 2005. Each cargo mission is followed by a piloted mission with a crew of four. The crew deploys, sets up, and puts into operation the support systems and scientific instrumentation delivered on that flight and on the preceding cargo flight. This approach was adopted to reduce the mass of the Earth/Moon orbital transfer vehicles and to allow for appropriate phasing of robotic and human activities for instrument and support systems emplacement.

Beginning in 2006, there will be one piloted lunar mission per year. It is expected that a return to the far-side observatory site will be required about every 3 years for instrument servicing, changeout, and upgrade. Other missions will explore selected near- and far-side sites.

Earth-to-orbit (ETO) transportation of equipment and propellants is accomplished by heavy-lift launch vehicles (HLLV's). The HLLV payload capability is assumed to be 91 t. Six HLLV launches are required during the 2-year setup phase. During the operational phase, two partial HLLV payloads are required for each annual servicing or exploration mission.

Assembly and mission crews are transported to the low Earth orbit (LEO) node and returned to Earth by the Shuttle. Crew transportation will be combined with other LEO node crew rotation requirements and has not been manifested separately.

Both cargo and piloted missions will use a lunar transfer vehicle (LTV) for transfer from the Earth to the Moon, insertion into lunar orbit, and return to Earth. A lunar descent vehicle (LDV) is used for both cargo and piloted landings on the lunar surface. Both the LTV and the LDV are chemically propelled; both are expendable. In addition, piloted missions use a lunar ascent vehicle (LAV) which also houses the crew during the transit from the Earth to the Moon and back.

Crew operations will take place during daylight only, limiting piloted missions to 14 days on the lunar surface. The crew will live out of the lander vehicle during their surface stay. Nominal mission duration from LEO departure to LEO return is about 20 days.

At the conclusion of each mission, the crew will return to the LEO node to await Shuttle pickup.

Aerobraking is used at Earth return to assist in achieving an orbit from which the crew can be retrieved and transferred to the LEO node.

The translunar trajectory permits a lunar flyby and return to LEO if abort is necessary before lunar orbit insertion.

2.3.1.3 Summary Data

Mission vehicles and payloads are assembled, checked out, and serviced at the LEO node. The node stores propellants delivered by the HLLV and transfers them to the mission vehicles. Maximum propellant load for all vehicles of a piloted mission is 96 t. Maximum propellant load for a cargo mission is 87 t.

The LEO node houses the mission crew of four prior departure for the Moon and after their return until they are picked up by the Shuttle for return to Earth. The node also houses the assembly/checkout crew of three.

The total mass through LEO is 246 t per year during the 2-year setup phase. During the operational phase, mass per year is 129 t. These masses can be reduced by about 10 percent if reusable space transportation vehicles are used.

Each cargo mission delivers 17.5 t, and each piloted mission, 6.5 t of user payloads and support equipment to the lunar surface. Landing accuracy is within 30 m of a designated point.

Capability is provided for twelve 6-hour EVA's by each of the four crew members on each piloted mission.

Assumed User Set. For purposes of analysis, table 2.3.1–I lists the types of equipment that would be provided at the observatory to accommodate user requirements.

Observatory Location. The observatory will be located on the far side of the Moon to eliminate electromagnetic interference from the Earth. The site will be on the lunar

TABLE 2.3.1-I.- LUNAR OBSERVATORY USER ACCOMMODATIONS

Lunar Satellites:
ComSats Imaging and cartographic polar orbiters Lunar monitoring orbiters
Lunar Exploration: Local (10-km) manned traverses Regional (1000-km) unmanned traverses Geophysical stations Unmanned sample collection
Astronomical Observation: Monitoring telescopes Optical telescope array Radio telescopes (very low frequency array, Moon-Earth interferometer, search for extraterrestrial intelligence) Solar observatory All-sky survey telescopes Other telescopes (coded-aperture gamma ray, X-ray, extreme ultraviolet, infrared)
Life Sciences: Man-tended, nonhuman life sciences laboratory
Support Equipment: Electrical power generation Construction equipment

equator to permit the entire sky to be viewed and also to allow return to Earth at any time without waiting for the orbit plane of the LTV to move to a favorable position.

The location in longitude has not been investigated in detail at this time because it is not a significant mission driver. To assure continuous contact between the communications satellites in halo orbits around the L2 point and both the Earth and the observatory, the site will have to be at least 5 degrees away from the near/far-side boundary. The exact dimensions of the halo orbit and any necessary antenna elevation above horizontal will increase this figure; the amount of increase has not been studied. Efficient operation of the MERI requires maximum effective distance between the Earth and the Moon; a site removed from the center of the far side achieves this without lines of sight too close to the local horizon.

2.3.2 Mission Definition and Manifest

Case Study 3 includes three piloted mission types (observatory construction, observatory maintenance, and exploration) plus the unmanned cargo missions. The payload manifests are generally different for each mission type, although some overlap occurs.

Table 2.3.2-I lists the payload items and the quantity of each item to be carried during each year of the 11-year

TABLE 2.3.2-I.- LUNAR OBSERVATORY – PAYLOAD ELEMENT MANIFEST

MSDB Number	Surface Systems Science Element Name	Mass kg	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14
LOM 001-A	Combined Imaging/	900		1													
LOM 005-A	Cartographic Lunar Polar Orbiter																
LOM 002-A	Lunar Polar Orbiter	1200	1														
LOM 003-C	Lunar Monitoring Orbiter	300					1	1	1	1							
LSM 001-C	Geophysical Station Network Mission	1210					4	4	4	4							
LSM 001-G	Geophysical Station Network	110						1	1	1		1	1		1	1	
LSM 008-C	Unmanned Sample Collection Mission	2110					2	2	2	2	2	2	2	2	2	2	2
LSM 008	Local Unmanned Traverse Mission	2865			1												
LSM 009	Regional Unmanned Traverse Mission	2865										1	1		1	1	
LSE 003-2	Portable Geohysical Exp. Package	52								1	1	1	1		1	1	
LSE 006	Geological Exploration Equipment	89								1	1	1	1		1	1	
LSE 009	Unpressurized, 10 km Manned Rover	550								1	1	1	1		1	1	
LSM 201	Optical Telescope (1m)	870					6										
LSM 202-1	Initial Very Low Freq. Radio Telescope	150					1										
LSM 203	Moon-Earth Radio Interferometer	2100					2										
LSM 204	Coded Aperture Gamma-Ray Telescope	2320						1									
LSM 205	X-Ray Telescope	2850						1									
LSM 206	Extreme Ultra-Violet Telescope	730						1									
LSM 211	Infrared Telescope	5700						1									
LSM 212	Radio Telescope	830					1										
LSM 213	Solar Observatory	2300					1										
LSM 217-1	Initial Monitoring Telescope	104					1										
LSM 217-2	Additional Monitoring Telescopes	10					9										
LSM 218	Visual All-Sky Survey Telescope	870						1									
LSM 220	X-Ray All-Sky Survey Telescope	2850						1									
None	Automated Biological Facility	4000						1									

MSDB Number	Surface Systems Support Element Name	Mass kg	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14
LOE 001-A	L2 Halo Comsat	2200	2														
NA	Power Unit	2000					1										
NA	Power Cable (6 km)	3000					1										
NA	Crane/Transporter	3800					1										
NA	Digger	1900					1										

MSDB Number	Transportation Systems Element Name	Mass (t)	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14
NA	Expendable Launch Vehicle	TBD	3	1	1												
NA	Lunar Transfer Vehicle (LTV)	13.2					2	2	1	1	1	1	1	1	1	1	1
NA	LTV Propellant - Piloted Mission	76.6					1	1	1	1	1	1	1	1	1	1	1
NA	LTV Propellant - Cargo Mission	74.6					1	1									
NA	Lunar Descent Vehicle - Piloted (LDV-P)	9.9					1	1	1	1	1	1	1	1	1	1	1
NA	LDV-P Propellant	13.0					1	1	1	1	1	1	1	1	1	1	1
NA	Lunar Ascent Vehicle - Piloted (LAV-P)	1.9					1	1	1	1	1	1	1	1	1	1	1
NA	LAV-P Propellant	2.0					1	1	1	1	1	1	1	1	1	1	1
NA	Lunar Descent Vehicle - Cargo (LDV-C)	4.0					1	1									
NA	LDV-C Propellant	12.8					1	1									

MSDB Number	Node Systems Element Name	Mass (t)	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14
NA	Space Station Vehicle Assembly Facility	30.0				1											
NA	Space Station Habitation Module	31.5			1												

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program assumed for this case study. In the years 2004 and 2005, the payloads listed will be divided as appropriate between the cargo mission and the piloted mission for that year. In subsequent years, all the payload items are carried on the single mission for the year.

Most of the scientific payloads were taken from the Mission and Supporting Elements Data Base (MSDB) developed by LESC. MSDB identification numbers are included for retrieval of more detailed information if desired. Masses given in the table are for one item, not the total per year.

The manifests shown in table 2.3.2-I do not account for the full payload capacity of the vehicles because the ERD/SRD payload capacities, which were based on rough preliminary estimates, exceed the actual payloads obtained by summing the assumed user set and associated surface systems. Subsequent iterations of this case study should attempt to eliminate this mismatch by reducing the vehicle payload capacity, adding more payload elements, or both. Improved mass estimates of the assumed payloads can also be expected to reduce the disparity.

2.3.3 Mission Architecture and Infrastructure

The overall mission architecture of the Lunar Observatory case Study is illustrated in figure 2.3.3-1. ETO transportation is accomplished by HLLV's. Assembly and mission crews are brought to the LEO node and returned by the Shuttle.

Viewed in comparison to the other case studies considered, the Lunar Observatory case makes relatively few demands on the supporting infrastructure. Maximum mass to LEO is about 250 t per year (see figure 2.3.3-2). However, that level is reached only during the 2-year setup phase of the program, in which two lunar missions are flown each year. During the operational phase, with a single mission per year, mass to LEO is about 130 t per year.

Flight crew personnel requirements are also relatively light, as shown in figure 2.3.3-2. The mission crew of four is required for only the 20-day duration of the flight plus any pre- and postmission time at the LEO node. The assembly and checkout crew of three are required for about 60 days per cargo or piloted mission.

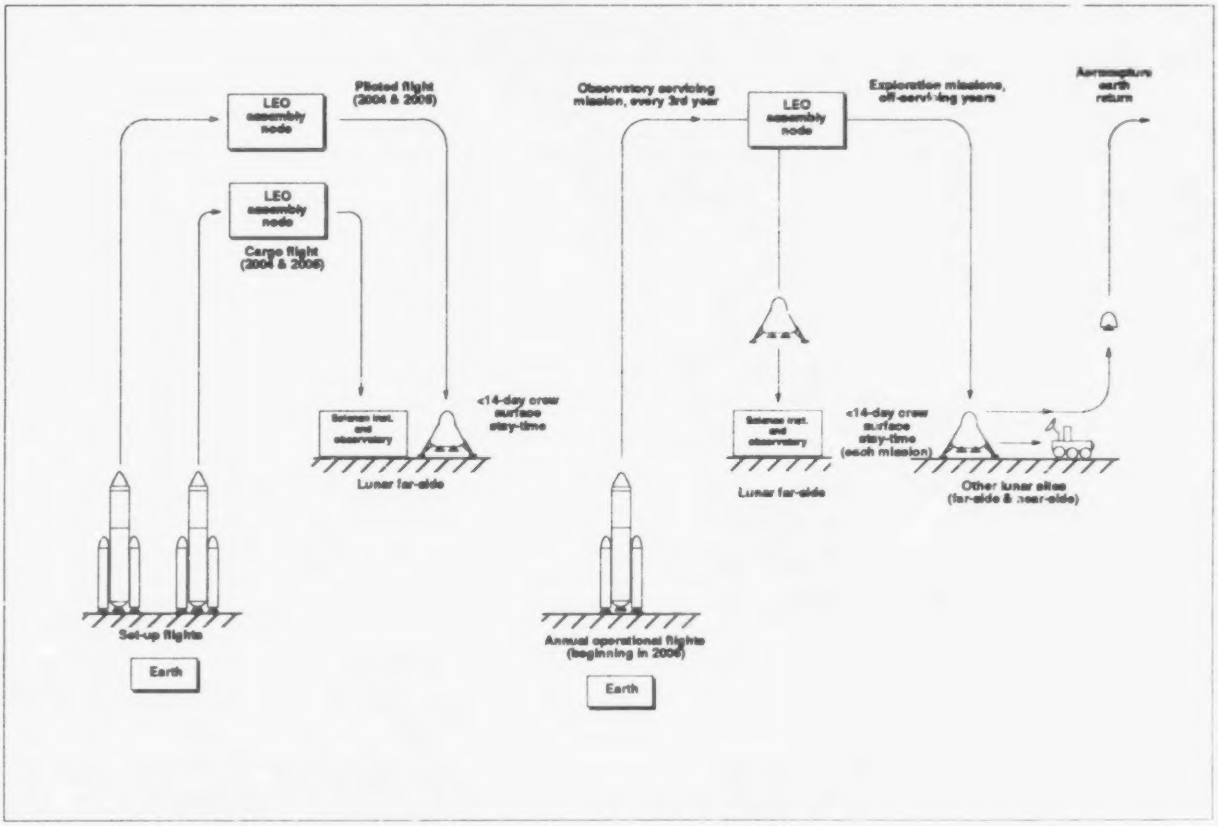
LEO assembly node
Piloted flight (2004 & 2006)
LEO assembly node
Cargo flight (2004 & 2006)
Science inst. and observatory
<14-day crew surface stay-time
Lunar far-side
Set-up flights
Earth
Observatory servicing mission, every 3rd year
LEO assembly node
Exploration missions, off-servicing years
Aerocapture earth return
Science inst. and observatory
<14-day crew surface stay-time (each mission)
Lunar far-side
Other lunar sites (far-side & near-side)
Annual operational flights (beginning in 2006)
Earth

Figure 2.3.3-1.- Lunar Observatory.

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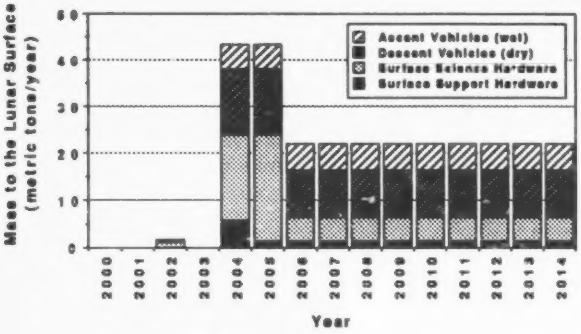
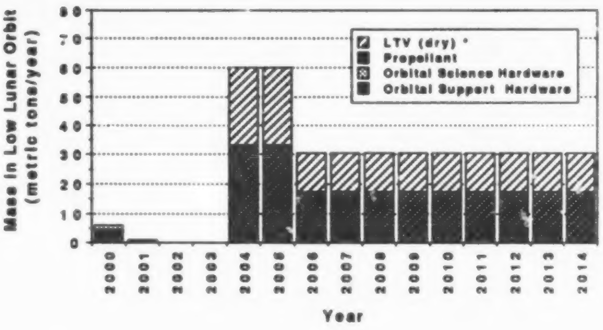
Lunar Observatory — Transportation Requirements



Notes: Mass to the Lunar Surface

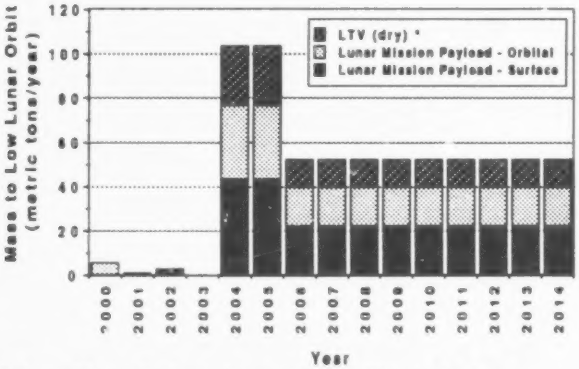
Ascent Vehicles (wet) includes the ascent stage of the piloted lunar descent vehicle and the propellant needed to launch it and the crew from the lunar surface to low lunar orbit where it can rendezvous with the waiting lunar transfer vehicle
Descent Vehicles (dry) includes the empty lunar descent vehicles, both piloted and cargo
Surface Science Hardware includes the equipment used to conduct the science experiments on the moon (e.g., telescopes, geophysical stations, geological exploration equipment, biology lab)
Surface Support Hardware includes the equipment used to support the science experiments on the moon (e.g., power systems, crane/transporter, equipment resupply)



Notes: Mass in Low Lunar Orbit (this includes the things that are used and/or remain in low lunar orbit)

LTV (dry) includes the empty lunar transfer vehicle
Propellant includes the return propellant used by the lunar transfer vehicles, and the propellant used by the lunar descent cargo and piloted vehicles
Orbital Science Hardware includes items such as a lunar polar orbiting satellite and a cartographic lunar polar orbiting satellite
Orbital Support Hardware includes communication satellites at L2

* The Lunar Transfer Vehicle (LTV) is a single stage vehicle. Therefore, its dry mass appears in both low lunar orbit and low Earth orbit



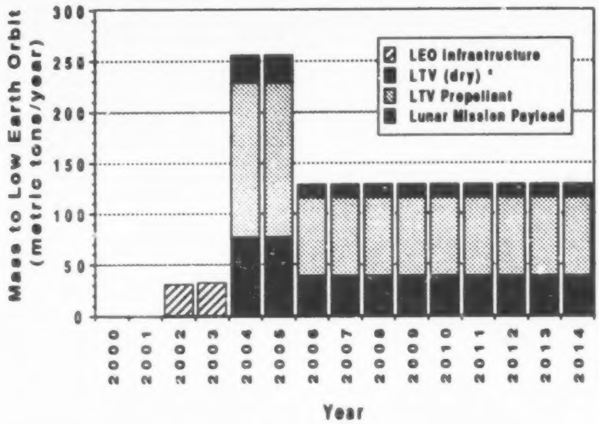
Notes: Mass to Low Lunar Orbit (this includes everything delivered to low Lunar orbit from low Earth orbit)

LTV (dry) includes the empty lunar transfer vehicle
Lunar Mission Payload - Orbital includes the return propellant used by the lunar transfer vehicles, the propellant used by the lunar descent cargo and piloted vehicles, the orbital science hardware, and the orbital support hardware

Lunar Mission Payload - Surface includes the dry piloted lunar descent vehicles, the fueled piloted lunar ascent vehicles, the dry cargo lunar descent vehicles, the surface science hardware, and the surface support hardware

Figure 2.3.3-2.- Lunar Observatory – transportation requirements.

Lunar Observatory-Transportation Requirements



Notes: Mass to Low Earth Orbit

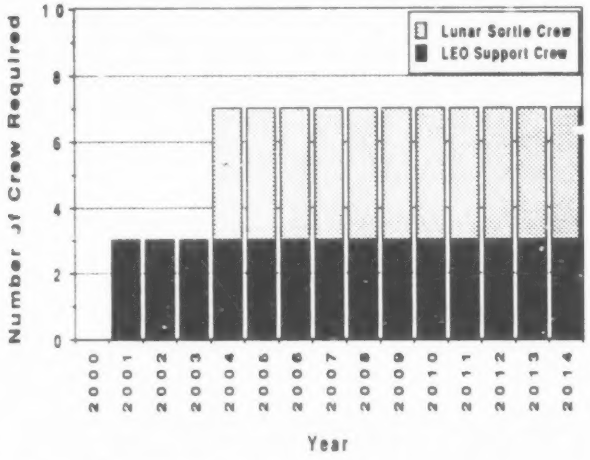
LEO Infrastructure includes space station facilities necessary to assemble and fuel the lunar transfer vehicles and their payloads and house the lunar sortie crews and the crews used to support the assembly operations

LTV (dry) includes the empty lunar transfer vehicle

LTV Propellant includes the propellant used by the lunar transfer vehicles in traveling from the earth to the Moon

Mission Hardware in LEO includes all science and support hardware, the fueled piloted lunar descent/ascent vehicles, and the fueled cargo lunar descent vehicles

- The Lunar Transfer Vehicle (LTV) is a single stage vehicle. Therefore, its dry mass appears in both low lunar orbit and low Earth orbit.



Notes: Number of Crew Required

Lunar Sortie Crew is the four-person mission crew that travels to the lunar surface for a 20 day total mission

LEO Support Crew is the number of crew needed in low Earth orbit to assemble, load, and fuel the lunar transfer vehicle

Figure 2.3.3-2.-Concluded.

Major program milestones are illustrated in figures 2.3.3-3 and 2.3.3-4.

2.3.4 Transportation Systems Definition

To set up an observational station on the lunar far side, several transportation elements will be required. These include a lunar transfer vehicle (LTV) to transfer humans and cargo to low lunar orbit (LLO), a lunar descent vehicle-piloted (LDV-P) to transfer the crew from LLO to the lunar surface, and a lunar descent vehicle-cargo (LDV-C) to transfer cargo from LLO to the lunar surface.

2.3.4.1 Elements and Systems Description

Reference System Description. In the baseline scenario, the LTV will depart from LEO with the LDV-C attached to the front of the aerobrake. When this vehicle reaches LLO, the LDV-C will detach from the LTV and make a propulsive descent to the lunar surface, while the LTV will remain in LLO.

The crew will then travel to the lunar surface in a similar fashion. The LTV will depart from LEO with the LDV-P attached to the front of the aerobrake. As before, when this vehicle reaches LLO, the LDV-P will detach and descend to the surface. After the crew has stayed for 14 days, they will ascend back into LLO in the LAV, which is a part of the LDV-P. The LAV will rendezvous and dock with the LTV, which will then transfer the crew back to LEO, using the Earth's atmosphere to aerocapture into LEO. The crew will rendezvous with the LEO node and return to the Earth's surface on the Space Shuttle. The missions using the LDV-C will occur only during observatory setup. All flights after that will use the LDV-P only. See figure 2.3.4-1 for a diagram of the trajectory.

Configuration and Mass Allocations. The LTV is shown in figures 2.3.4-2 and 2.3.4-3. Like all vehicles in this case study, it uses LOX/LH2 propellants and RL-10-derivative engines. This vehicle consists of four propellant tanks, four engines, an aerobrake, and avionics. A propellant tankage factor of 10 percent and zero boiloff were assumed in sizing the tanks of all of the vehicles in this case study. Attachment points on the front of the aerobrake will be used to attach both the LDV-P and the LDV-C.

The LDV-C is shown in figures 2.3.4-4 and 2.3.4-5. The vehicle is essentially a large cargo bay with a volume capacity of 370 m^3. The LDV-C carries 17.5 t of payload to the lunar surface by using four RL-10-derivative engines and LOX/LH2 propellant. The diameter of the cylindrical cargo bay is 7.6 m with a height of 10 m. The descent can still be accomplished if one engine fails, by turning off the engine on the opposite side of the failed engine.

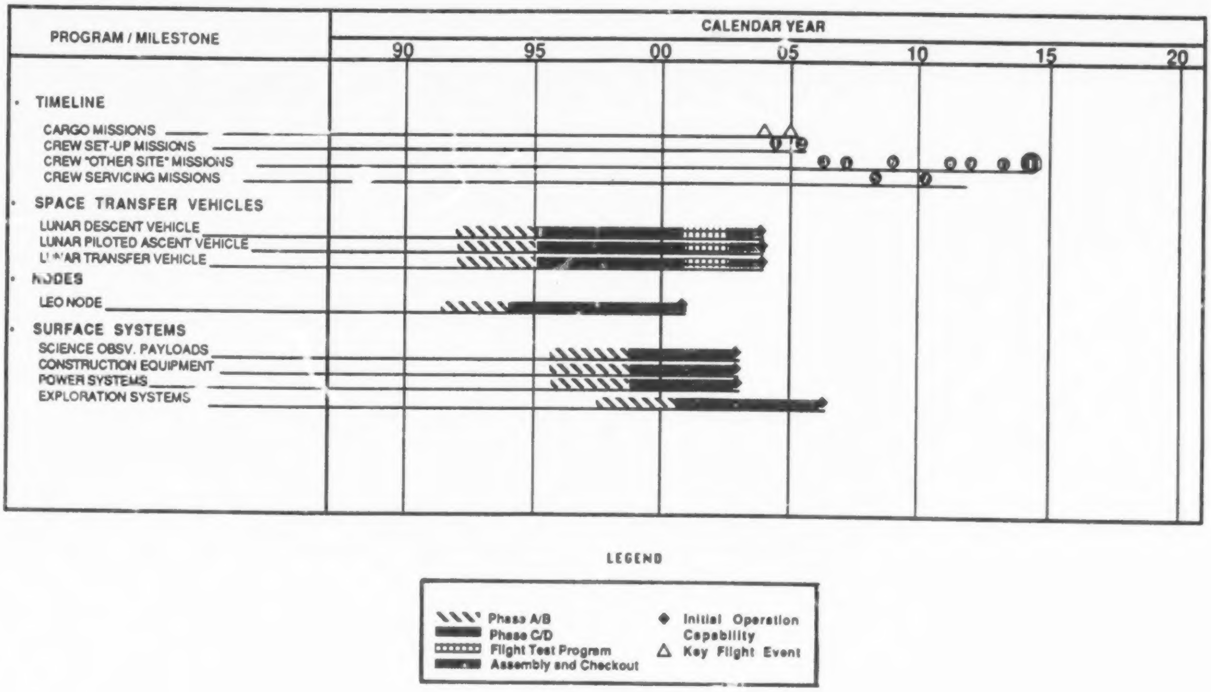


Figure 2.3.3-3.- Milestones for lunar observatory --mission requirements.

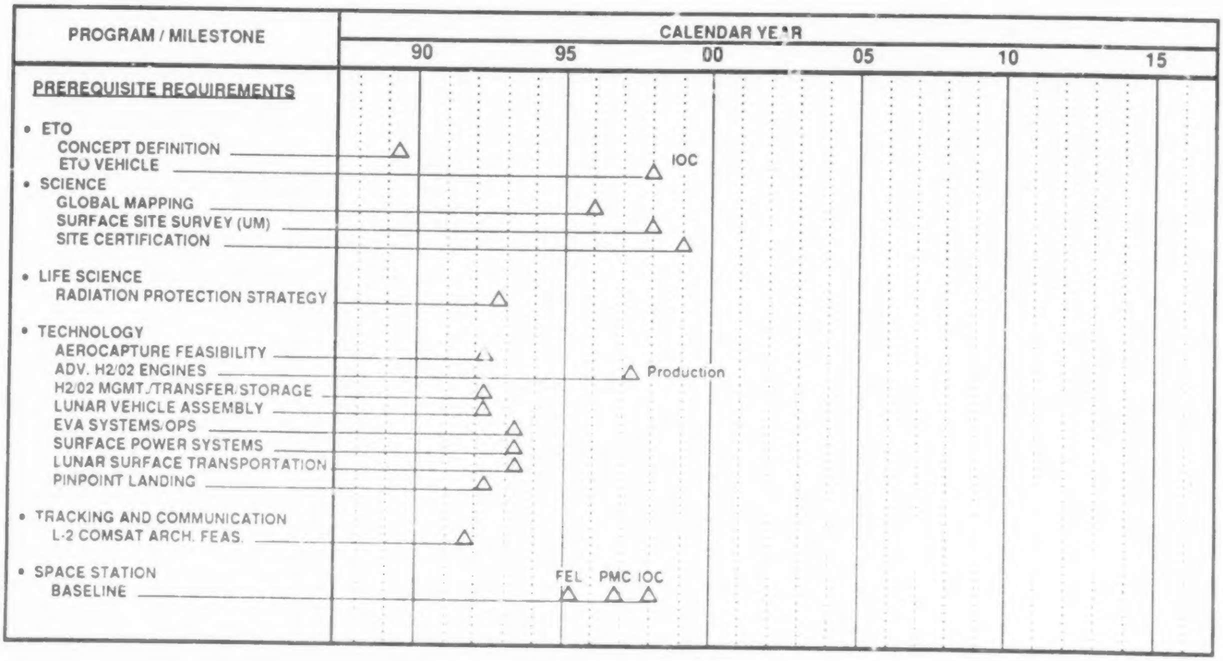


Figure 2.3.3-4.- Milestones for lunar observatory --prerequisite requirements.

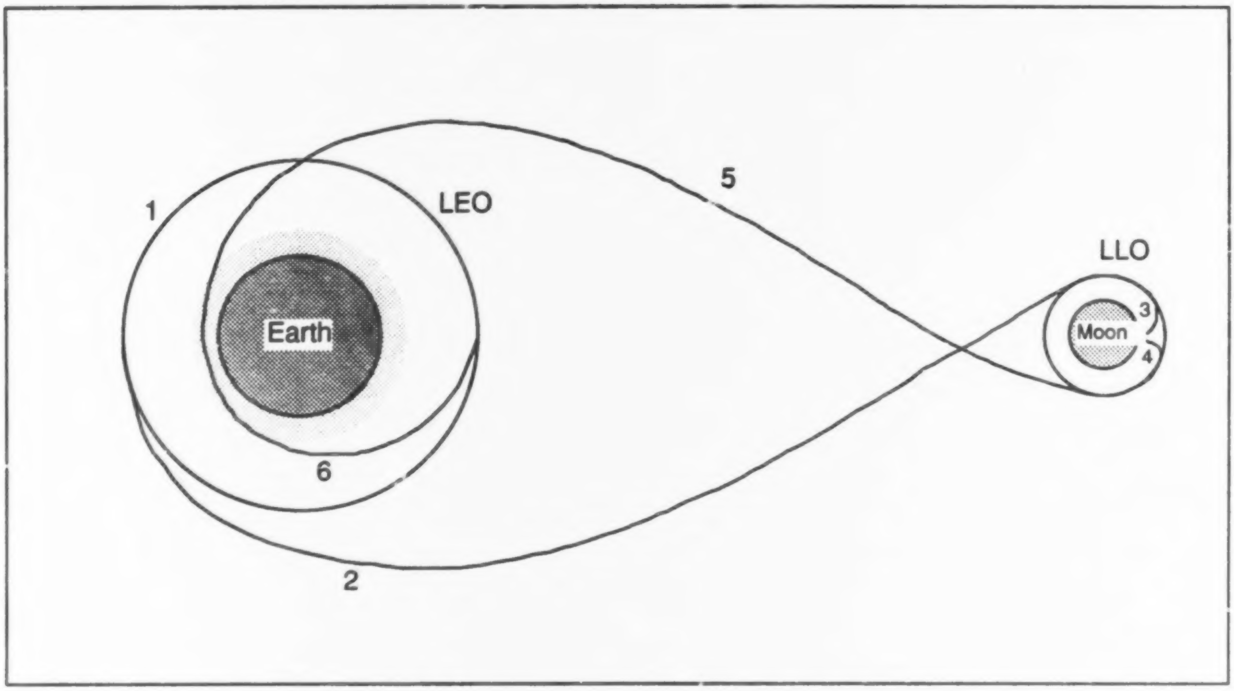


Figure 2.3.4-1.- Case Study 3 – lunar observatory missions.

Dry Mass	13,244 kg
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Thrust (total)	392 kN (88.1 klb,)
I_{sp} (460 sec)	4.51 kN-s/kg
Max. Propellant Capacity	80,859 kg
Tank Mass	8,086 kg
Total Mass	94,103 kg

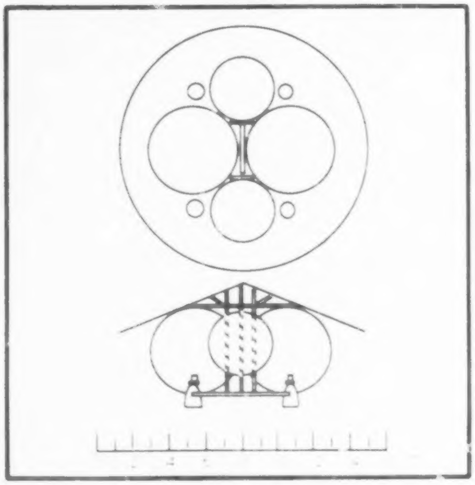


Figure 2.3.4-2.- Lunar transfer vehicle (LTV) with mass breakdown.

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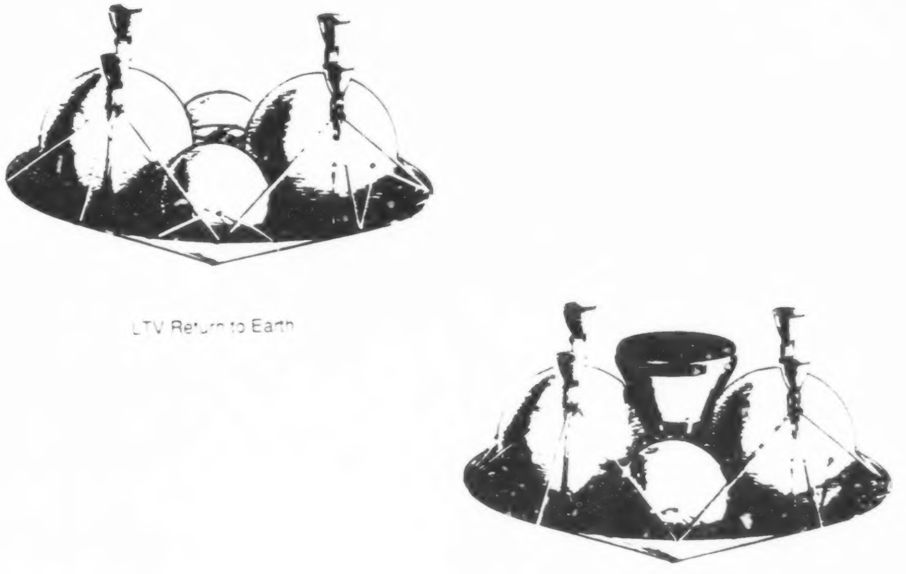

LTV Return to Earth

Crew Return to Earth

Figure 2.3.4-3.-Lunar transfer vehicle (LTV), artist's rendition.

Dry Mass	
(includes payload)	21,519 kg
Payload Mass	
(user payload)	17,500 kg
Payload Volume	
Cargo Bay	
(cyl.- 7.6m dia., 8.1m ht.)	370 m³
Propulsion System	
Propellant Type	LOX/LH_2
Engines	
Number	4
Type	RL10B-2
Mass (ea.)	191 kg
Max. Thrust (total)	392 kN (88.1 klb,)
Isp (460 sec)	4.51 kN-skg
Propellant Mass	12,768 kg
Tank Mass	1,277 kg
Initial T/W_L	7.0 @ 100% throttle
Final T/W_L	11.2 @ 100% throttle
Mass Fraction	
Descent from LLO	0.628

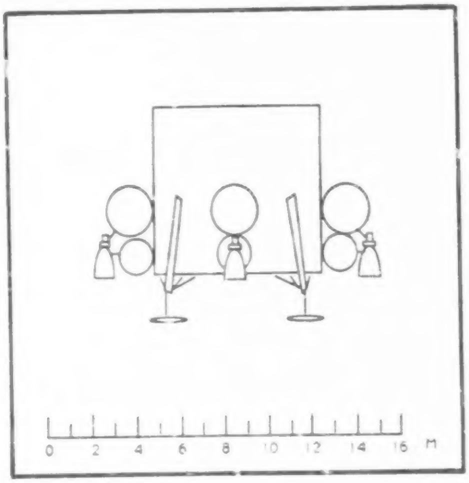

Figure 2.3.4-4.- Lunar descent vehicle-cargo (LDV-C) with mass breakdown.

The LDV-C employs an elevator to lower the cargo to the lunar surface from the cargo bay, negating the need for cranes and other unloading equipment and thus minimizing the surface infrastructure required. This translates directly to a savings in mass in LEO.

The LDV-P is shown in figure 2.3.4-6. It is based on the Mars descent Vehicle (MDV) in Case Study 2. The bottom level of the LDV-P is a 7.6-m-diameter, 3-m-high habitation module where the crew will live for the 14-day staytime. A 55-day safe haven capability is assumed in addition to the 14-day nominal staytime. Braced on top of the module is the LAV which carries the crew back to LLO. The LDV-P makes a descent to the lunar surface using four RL-10-derivative engines and LOX/LH2 propellant. Again, the descent can be made if one engine fails, by turning off the engine on the opposite side of the failed engine.

The LAV shown in figure 2.3.4-7 is based on the Mars ascent vehicle (MAV) in Case Study 2. The LAV is a Gemini-based, lightweight capsule which carries the four crewmembers from the lunar surface to LLO for rendezvous with the LTV waiting in orbit. It uses RL-10-derivative engines and a total thrust level lower than those used for the LTV, LDV-P, and LDV-C. This vehicle allows a sufficient thrust level for an engine-out capability.

ETO, Onorbit Assembly, and Servicing Needs. Assuming an HLLV capability of 91 t, a minimum of six launches will be required initially to set up the lunar observatory during the years 2004 and 2005. After the initial setup, only two HLLV launches a year will be required to maintain the observatory and investigate new sites on the lunar surface.

The HLLV lift capability will allow most of the transportation elements to be sent up in a single launch, thus requiring no onorbit assembly. The LDV-C's and LDV-P's will be launched into LEO fully assembled and fueled. The LTV's will be sent up fully assembled also, but they will not be fueled.

Cryogenic propellant will be sent on a separate launch and in-space propellant transfer will be necessary to fuel the LTV's. The LDV-P's and the LDV-C's will also need to be attached to the LTV's in orbit. This can be accomplished telerobotically, or by the crew of the LEO node. STS flights will be necessary to bring up the crew from the Earth's surface at the start of the mission and to bring the crew back down to the surface at the end of the mission.

Transportation Program Development Schedule. The schedule for development and flight-testing of the transportation system and the schedule for required precursors is shown in figure 2.3.4-8.

Precursor Missions. Communication satellites will need to be set up at L2. A lunar polar orbiter, an imaging/cartographic lunar orbiter, and an unmanned local traverse mission (LTM) will need to determine the location of the site for the lunar observatory, and other sites of interest for future missions.

2.3.4.2 Enabling Technology Needs

Propulsion System. Increased performance of the RL-10-derivative engines must be verified. The feasibility of small, very low boiloff cryogenic propellant tanks must be evaluated.

Aerobrake Technology. The feasibility of using an aerobrake to capture the LTV into LEO must be evaluated. An aerobrake must be designed and flight-tested. If an effective aerobrake cannot be designed, propulsive braking options will have to be considered. Also, the ability to attach the LDV-P's and the LDV-C's to the front of the aerobrake using ceramic hardpoints must be verified.

2.3.4.3 System Alternatives and Opportunities

One important tradeoff evaluated was reusability of the LTV's. Because they must be refueled in orbit there is no real benefit in discarding them. Refurbishing and refueling LTV's rather than launching a new vehicle for every mission would save almost 14 t per mission. Although reuse of cargo mission LTV's would require an additional 11 t of propellant for return to LEO, that mass penalty would be more than balanced by the mass saved by not launching new LTV's. There is also a manufacturing cost saving, since fewer LTV's need be produced. Finally, discarding LTV's would result in a significant buildup of residual hardware in LEO and LLO, resulting in hazardous orbiting conditions.

One major alternative to the baseline lunar observatory mission scenario would be to eliminate the stopover of the space vehicles in LLO and in LEO. A direct descent to the lunar surface and a direct Earth return would afford a mass savings of over 17 percent and would allow the astronauts to return directly to the Earth's surface, rather than relying on an STS flight. A summary of the alternatives along with a mass comparison is shown in table 2.3.4-II.

Direct reentry at Earth could also be used in either the baseline scenario or in the recoverable LTV option. In both of these cases, the crew would separate from the LTV and descend directly to the surface. This would eliminate the necessity of rendezvous with the LEO node and save a Shuttle mission that would be needed to recover the astronauts. The LTV could still be aerobraked into LEO and be recovered for refurbishment and refueling.

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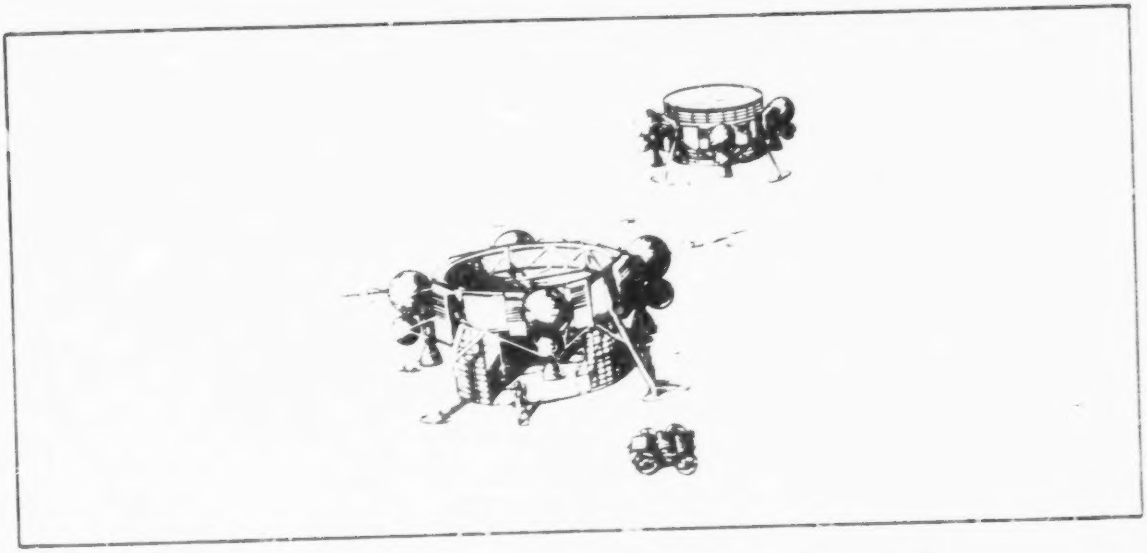

Figure 2.3.4-5.-Lunar descent vehicle-cargo (LDV-C), artist's rendition.

Dry Mass (includes payload)	21,866 kg
Payload Mass	
(crew equipment & supplies)	8,044 kg
Payload Volume	
LAV	
(cone - 3.6m dia, 2.3m ht)	8 m³
HAB Mod	
(cyl - 7.6m dia, 3m ht)	136 m³
Propulsion System	
Propellant Type	LOX/LH
Engines	
Number	4
Type	RL10B-2
Mass (ea)	191 kg
Thrust (total)	392 kN (88.1 klb)
I_{sp} (460 sec)	4.51 kN-s/kg
Propellant Mass	12,974 kg
Tank Mass	1,297 kg
Initial T/W	6.9 @ 100% throttle
Final T/W	11.0 @ 100% throttle
Mass Fraction	
Descent from LLO	0.628
Total Mass	34,840 kg

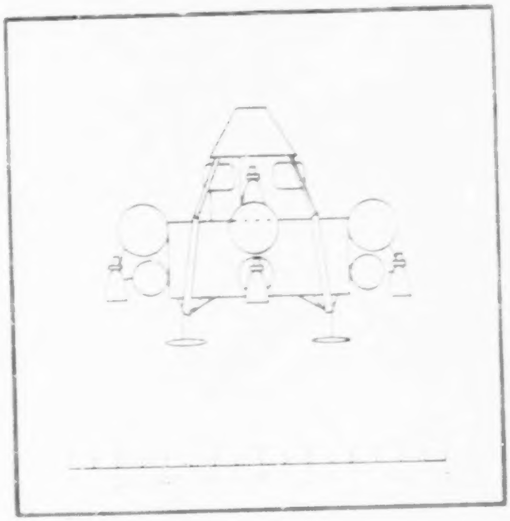

Figure 2.3.4-6.- Lunar descent vehicle-piloted (LDV-P) with mass breakdown.

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Dry Mass (includes payload)	439,140 kg
Payload Mass (MTV)	334,130 kg
Propulsion System	
Propellant Type	LOX/LH$_2$
Engines	
Number	4
Type	SSME - derivative
Mass (ea.)	3,175 kg
Thrust (total)	2,415 kN
I_{sp} (480 sec)	4.71 kN - s/kg
Propellant Mass	685,3330 kg
Tank Mass	21,100 kg
Initial T/W	0.22
Mass Fraction	
Total	0.39
Total Mass	1,124,470 kg

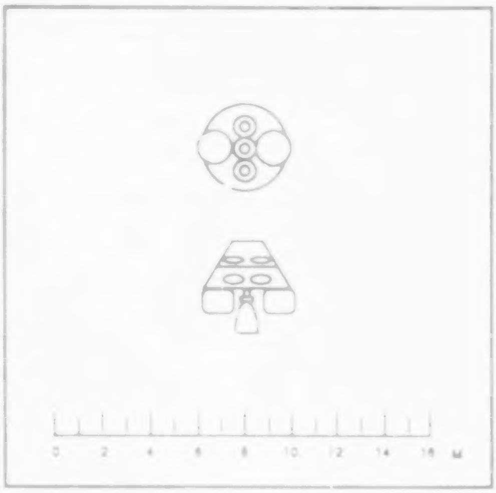

Figure 2.3.4-7.- Lunar ascent vehicle (LAV) with mass breakdown.

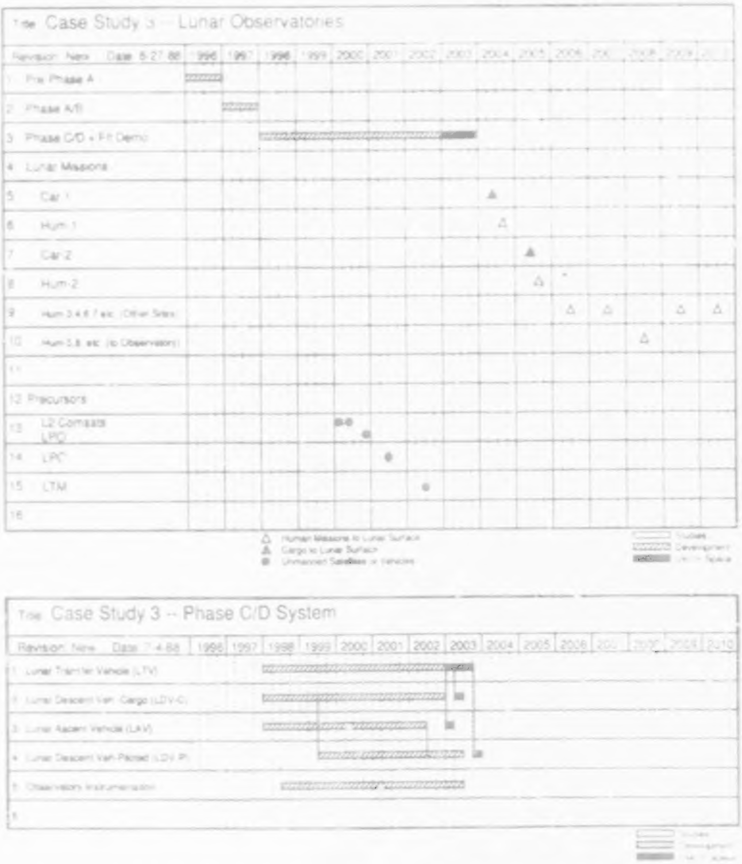

Figure 2.3.4-8.- Transportation program development schedule.

TABLE 2.3.4-1 - LAUNCH SUMMARY FOR VARIOUS OPTIONS

Year	'00	'01	02	'03	'04
No. of Launches	3	3	2	2	2
IMLEO(t)					
Baseline	246	246	129	129	129
Direct entry	212	212	115	115	115
Reuseable LTV*	242	228	116	116	116

*Returns LTV used for cargo as well as LTV/LAV

2.3.5 Orbital Node Systems Definition

2.3.5.1 LEO Orbital Node Elements and System Description

The Lunar Observatory is an 11-year operational program (FY 1988 study constraint only) which requires a transportation node in LEO. This case study is characterized by a single cargo mission and a single piloted mission in 2004 and 2005, followed by one piloted mission per year through 2014. Staytimes at the lunar surface are 14 days. The short trip times mean that a zero-g countermeasures program is not required for this case study.

Because of the much smaller dry masses associated with the lunar mission, the entire vehicle can be brought to orbit on a single HLLV launch. This eliminates the need for onorbit assembly if expendable vehicles are used. Some degree of onorbit checkout will be required, however, and the vehicles must still be fueled onorbit, requiring the capability for propellant storage and handling at the LEO transportation node. If reusable vehicles are employed, onorbit servicing and assembly will be required. In addition, these systems will return to the station with some residual propellants. This will create the need for a wet/hazardous processing facility at the station.

For a given flight, rough timelines for the cargo and piloted missions are shown in figures 2.3.5-1 and 2.3.5-2. In these figures LS refers to lunar surface and LLO refers to low lunar orbit. With the assumption of six HLLV flights per year, 75 to 90 days of operations at the transportation node are required to support the lunar missions. As shown in the figures, a single HLLV delivers the space vehicle and a partial fuel load. The second HLLV delivers the remainder of the fuel. For the piloted vehicles, the lunar crew is transported to the node by the STS, spends 14 days on the surface of the Moon, returns to the node using aerobraking and pickup by the OMV, then returns to Earth via the STS.

Figures 2.3.5-3 and 2.3.5-4 show the additions to Space Station Freedom required for it to become a transportation node. These are based on the following operations scenario:

a. OMV retrieval of HLLV payload (dry vehicles)

b. Vehicle assembly (if required)

c. Checkout of all systems

d. OMV retrieval of HLLV payload (propellants)

e. Propellant transfer to lunar vehicle

f. Prestaging checkout

g. OMV transfer of lunar vehicle to departure point

h. OMV retrieval of lunar vehicle element (if resuable)

i. Service/refurbishing of vehicle (if resuable)

j. OMV staging of lunar crew return vehicle for disposal (if expendable)

With an HLLV launch frequency of six per year, the total processing, mission operations, and recovery can be accomplished in less than 90 days. Thus, even in the first two years of operations when there are two flights to the lunar surface, the Space Station Freedom research and development operations would be impacted for less than 180 days. It also appears that a three-person, single-shift operation is adequate for accommodation of Case Study 3.

Facilities required at the station are an unpressurized hangar sized to handle the lunar vehicles and OMV's and lunar and OMV propellant storage facilities. The hangar facility is sized at 30 m by 15 m by 15 m. The mass of the hangar and associated equipment is 15 t. The propellant storage facility (tanks) mass is 17 t.

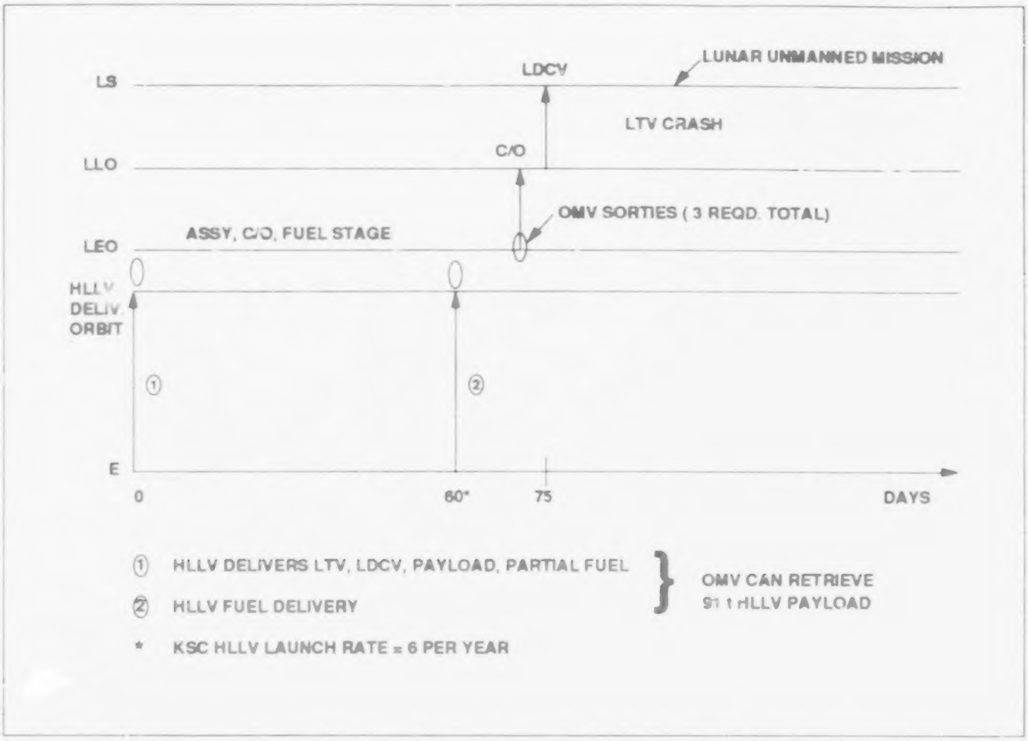
LUNAR UNMANNED MISSION
LS
LDCV
LTV CRASH
C/O
LLO
OMV SORTIES (3 REQD. TOTAL)
ASSY, C/O, FUEL STAGE
LEO
HLLV DELIV. ORBIT
①
②
E
0
60*
75
DAYS
① HLLV DELIVERS LTV, LDCV, PAYLOAD, PARTIAL FUEL
② HLLV FUEL DELIVERY
* KSC HLLV LAUNCH RATE = 6 PER YEAR
OMV CAN RETRIEVE 91 t HLLV PAYLOAD

Figure 2.3.5-1.- Case Study 3 -- cargo mission timeline.

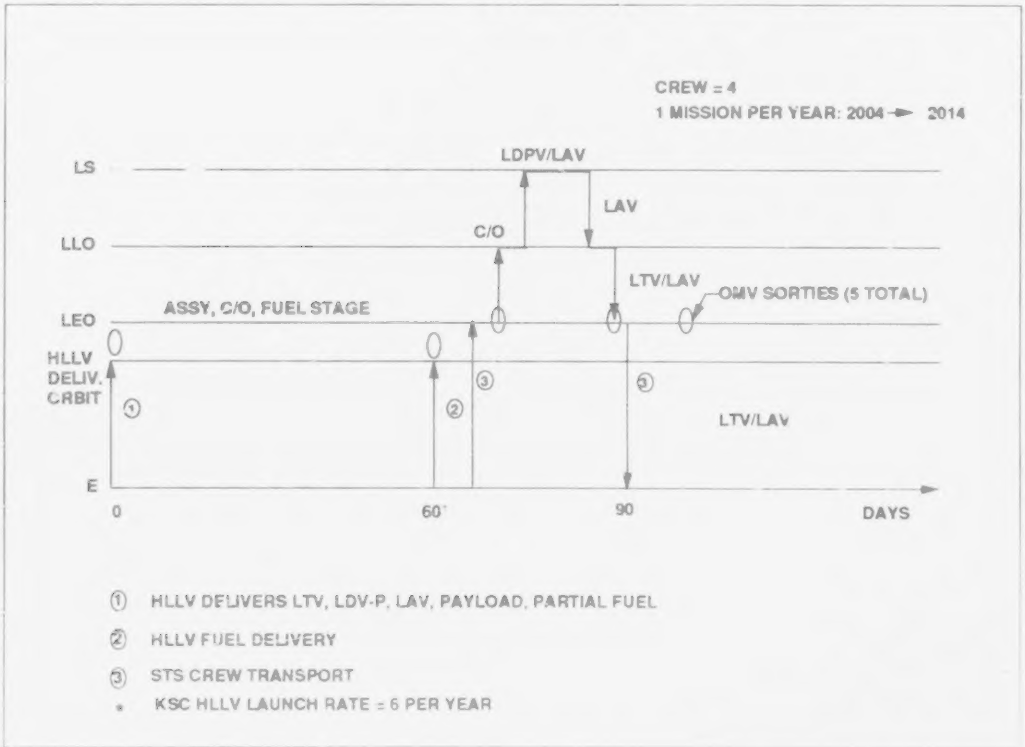
CREW = 4
1 MISSION PER YEAR: 2004 → 2014
LS
LDPV/LAV
LAV
C/O
LLO
LTV/LAV
OMV SORTIES (5 TOTAL)
ASSY, C/O, FUEL STAGE
LEO
HLLV DELIV. ORBIT
①
②
③
③
LTV/LAV
E
0
60*
90
DAYS
① HLLV DELIVERS LTV, LDV-P, LAV, PAYLOAD, PARTIAL FUEL
② HLLV FUEL DELIVERY
③ STS CREW TRANSPORT
* KSC HLLV LAUNCH RATE = 6 PER YEAR

Figure 2.3.5-2.- Case Study 3 -- piloted mission timeline.

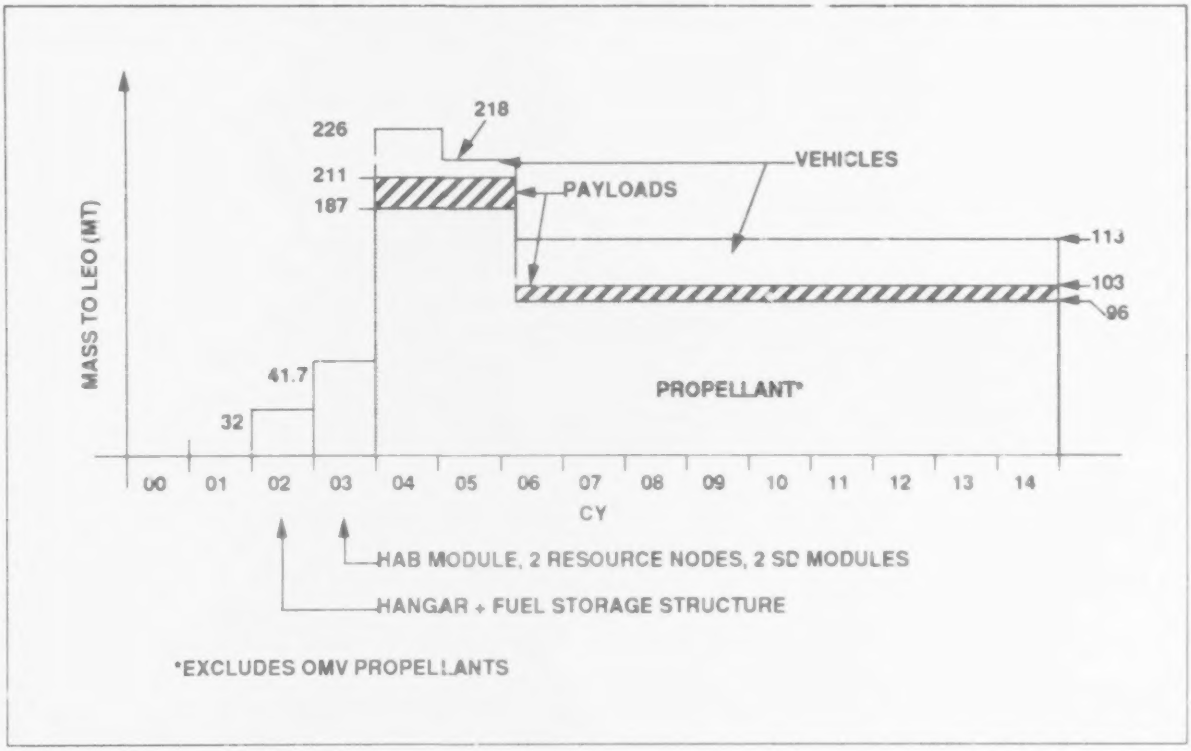

Figure 2.3.5-3.- Case Study 3 -- mass to LEO (reusable LTV, LAV).

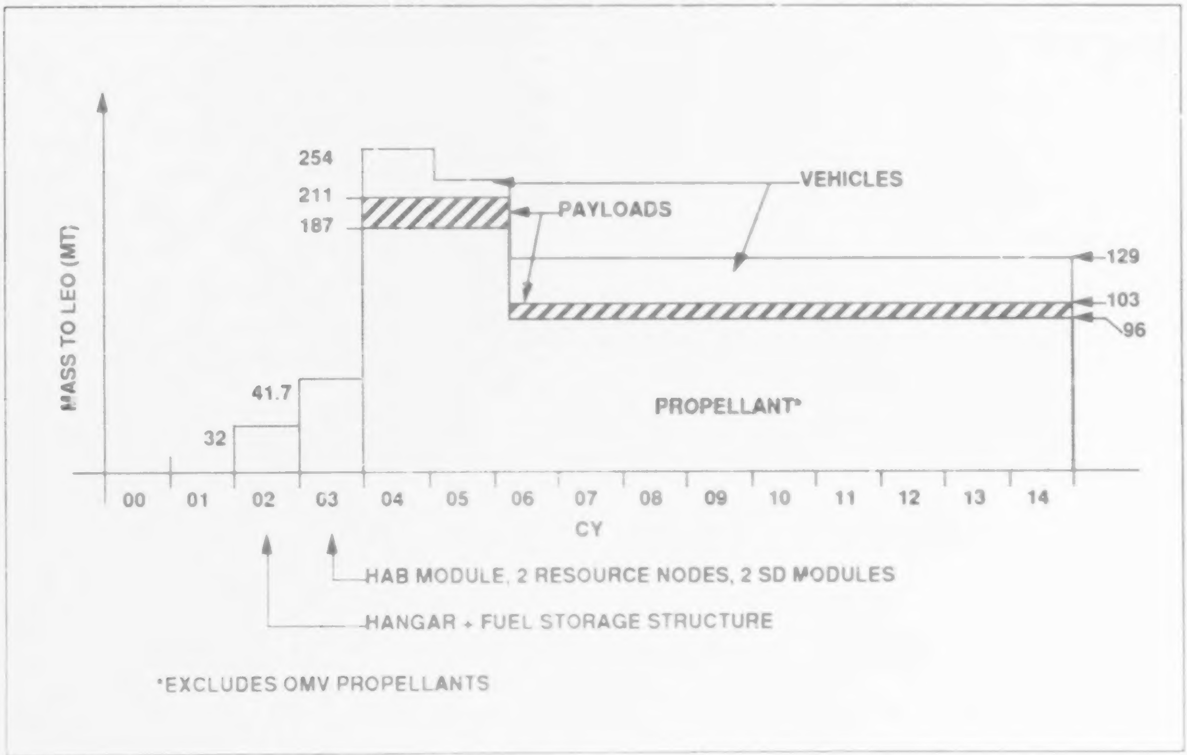

Figure 2.3.5-4.- Case Study 3 -- mass to LEO (expendable vehicles).

An additional hab module is required to accommodate the assembly crew plus the transient lunar crews. Two additional nodes are required to provide access to other station elements and dual egress.

Mass summaries of two concepts for Space Station Freedom accommodation of the lunar mission are given in figure 2.3.5-5. The two configurations studied are illustrated in figures 2.3.5-6 and 2.3.5-7. Concept A is minimal additions to the phase 1 station. The hangar is located at the "scar" point for the servicing facility (a Space Station Freedom phase 2 element). Total additional hardware is 175.6 t, most of which is propellant and tanks and the 2 OMV's. An additional hab module has been added to accommodate the assembly crew as well as transient crews on their way to the Moon. Concept C uses the dual keel configuration and locates the hangar and propellant storage facilities to minimize movement of the station center of mass. This concept minimizes the impact of lunar operations on the station microgravity research program (during quiescent periods). It requires a mass increase of about 20 t for truss bays and utility runs with propellant and lunar vehicles at the station. The total mass in orbit is about twice the mass of the phase 1 station. Choice of either of the two concepts as the preferred option requires substantially more analysis, particularly in terms of the specific operations involved in preparation and checkout of the lunar vehicles.

2.3.5.2 Technology Drivers

If the lunar observatory mission uses expendable vehicles, the primary technology driver for the transportation node function is cryogenic propellant storage and handling. Automated rendezvous and docking are required for OMV pickup of the HLLV payload and return to the transportation node. If reusable vehicles are employed to achieve the ETO mass savings described in the previous section, onorbit servicing of the LTV and LAV will be required, including the requirement for cryogenic propellant storage and handling. These vehicles will have some residual propellants and are classified as "wet" or hazardous from the processing point of view. The degree to which the aeroshells must be refurbished is unknown at this time, but could require some technology advances. Certainly, the ability to process the hazardous vehicles onorbit will require advances in automation and telerobotics.

	CONFIGURATION			
COMPONENT	CONCEPT A		CONCEPT C	
	NUMBER	MASS (Kg)	NUMBER	MASS (Kg)
TRUSS BAYS	8	750	54	3,900
UTILITY BAYS	8	250	50	15,800
SOLAR DYNAMICS	2	11,400	2	11,400
HABITAT MODULE	1	19,500	1	19,500
NODE	2	8,700	2	8,700
PROPELLANT + TANKS	6	113,000	6	113,000
HANGER + EQUIPMENT	1	15,000	1	15,000
OMV (DRY)	2	7,000	2	7,000
TOTAL ADDITIONAL HARDWARE		175,600		194,300
+ PHASE 1 STATION 219,000 Kg.		394,600		413,300
+ LUNAR CARGO VEHICLE 34,700 Kg.	1	429,300	1	448,000
OR + LUNAR PILOTED VEHICLE 32,800 Kg.	1	427,400	1	446,100

Figure 2.3.5-5.- Case Study 3 -- Space Station Freedom growth hardware to accommodate lunar observatory

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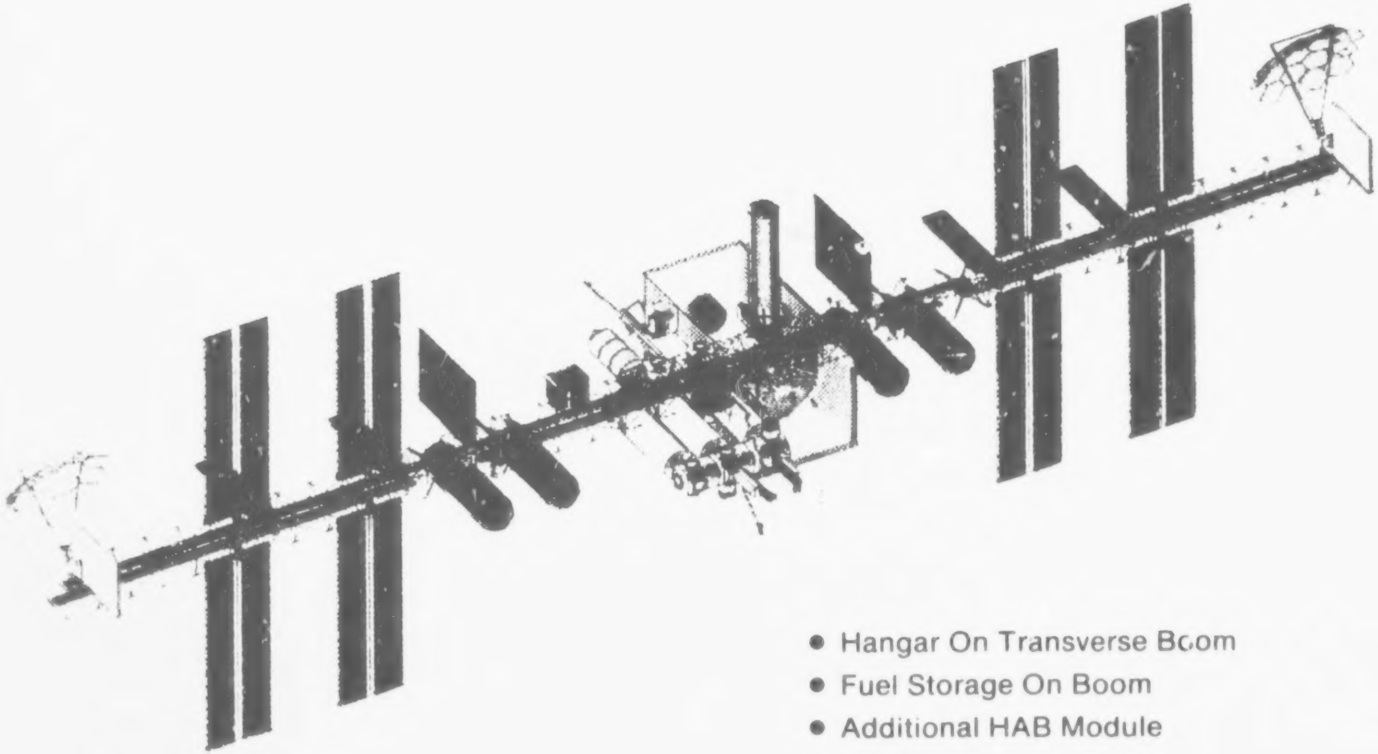
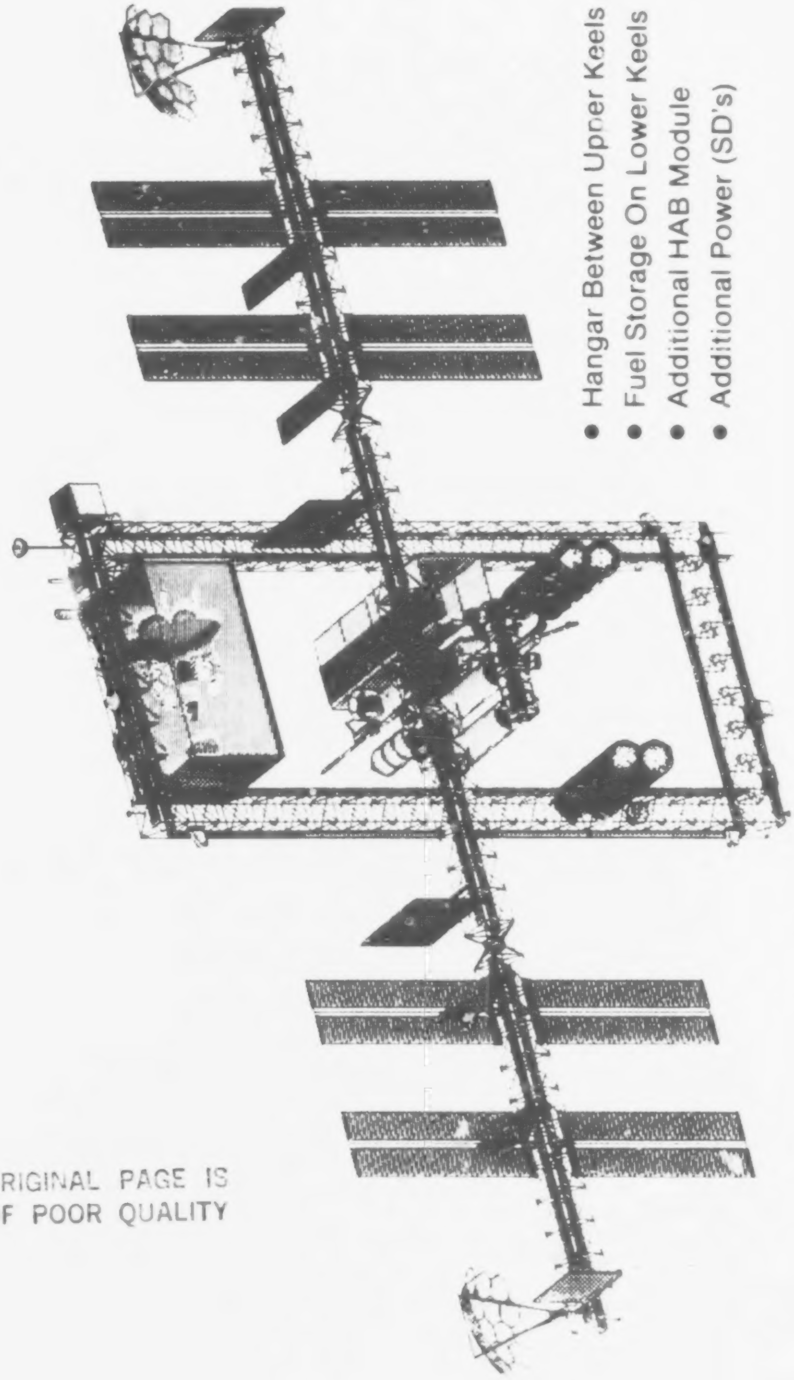


- Hangar On Transverse Boom
- Fuel Storage On Boom
- Additional HAB Module
- Additional Power (SD's)

Figure 2.3.5-6.- Concept A lunar observatory accommodation.



- Hangar Between Upper Keels
- Fuel Storage On Lower Keels
- Additional HAB Module
- Additional Power (SD's)

Figure 2.3.5-7.- Concept C (centered) lunar observatory accommodation.

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Figure 2.3.5-7. Concept C (centered) lunar observatory accommodation.

2.3.5.3 System Alternatives and Opportunities

A comparison of mass to LEO requirements for expendable and reusable vehicles is given in figures 2.3.5-3 and 2.3.5-4. If we assume the LTV and the LAV to be viable candidates for reuse, mass savings of 28 t in 2004, 36 t in 2005, and 16 t per year thereafter are achievable, though at the cost of onorbit servicing of the LTV and LAV. Further study is required to determine whether the cost is greater than the savings.

2.3.6 Planetary Surface Systems Description

The purpose of Case Study 3 is to construct and assemble a scientific laboratory and astronomical observatory on the Moon. These facilities will provide capabilities greater than current ones on or near Earth. Table 2.3.6-I summarizes the lunar observatory improvements in science data return over current capabilities.

2.3.6.1 Elements and Systems Description

Figure 2.3.6-1 summarizes the surface elements identified to support Case Study 3. In particular, the list includes five systems: the lunar surface EVA systems, a power system, an unpressurized rover, construction equipment, and a regolith bagger to be used to provide emergency radiation protection.

Figure 2.3.6-2 depicts the layout concept for the far side observatory. The assumed surface science set that follows totals 40 metric tons in first 2 years.

a. Very low frequency radio array

b. Optical very large array

c. Stellar monitoring, telescopes (optical, X-ray, gamma ray, ultraviolet, and infrared)

d. Moon-Earth radio interferometer

e. Solar observatory

f. Radio telemetry for SETI

g. Local geological traverses in unpressurized rover

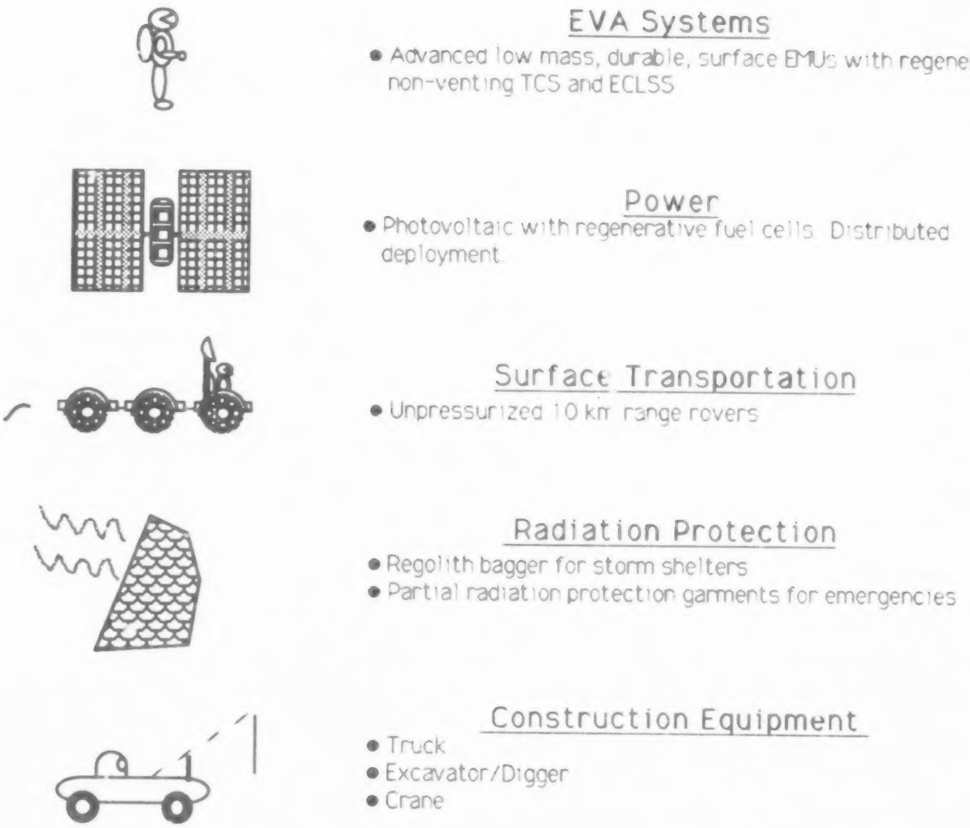
h. Geophysical stations

A very broad spectrum of frequencies can be covered with this observatory, from very low to very high. The far side of the Moon provides a unique place for observatories, such as the very low frequency array, because of shielding from Earth noise. The Moon is a stable platform for optical telescopes, requires less pointing accuracy than orbiting facilities, and provides a 14-day "night" to observe stars. Because of its slow rate of sidereal rotation, the Moon provides a unique platform for studying longer period variations in stars and other bodies. Hence, a tremendous amount of scientific data can be acquired in this case study.

Construction and service requirements for the various observatory instruments are presently not well defined

TABLE 2.3.6-I.- LUNAR OBSERVATORY SCIENCE CAPABILITIES

OPPORTUNITY	CAPABILITY
Optical Very Large Array	10^4 improvement in resolution over HST
Moon-Earth Radio Interferometer	10^3 improvement in resolution over VLA
Very Low Frequency Radio Array	Currently unexplored spectral region
Geological Exploration	Early lunar evolution; history of solar radiation; impact processes; etc.



EVA Systems

- Advanced low mass, durable, surface EMUs with regenerable, non-venting TCS and ECLSS

Power

- Photovoltaic with regenerative fuel cells. Distributed deployment.

Surface Transportation

- Unpressurized 10 km range rovers

Radiation Protection

- Regolith bagger for storm shelters
- Partial radiation protection garments for emergencies

Construction Equipment

- Truck
- Excavator/Digger
- Crane

Figure 2.3.6-1.- Planetary surface systems elements for lunar observatories.

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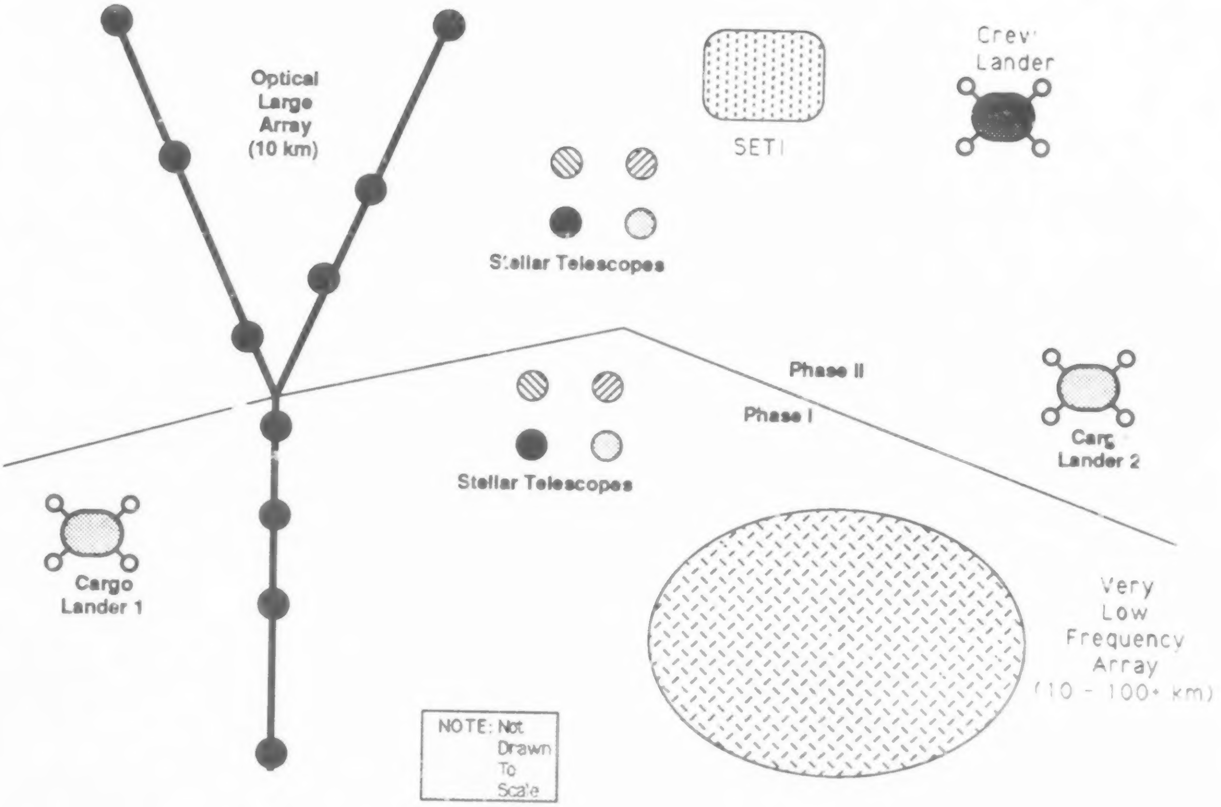
Optical
Large
Array
(10 km)
SETI
Crew
Lander
Stellar Telescopes
Phase II
Phase I
Stellar Telescopes
Cargo
Lander 2
Cargo
Lander 1
Very
Low
Frequency
Array
(10 - 100+ km)
NOTE: Not
Drawn
To
Scale

Figure 2.3.6-2.- Lunar observatory layout.

Construction and service requirements for the various observatory instruments are presently not well defined. Because the payloads and operations are quite diverse, automation opportunities would likely be limited to generic, repetitious, or high-leverage tasks; e.g., site clearing or deploying elements in a large array. Available preliminary estimates indicate manpower needs may be met with the baseline EVA provisions: four crew working 6 hours a day for 12 days during each construction mission. Note that 6 hours of productive EVA will probably entail more than 10 hours of crewtime to allow for preparation, meals, cleanup, etc. This time estimate is based on the following assumptions. Typical construction tasks would involve unloading and transporting payloads, preparing sites, and deploying payloads. Site preparation would be minimal and consist primarily of surveying, clearing, and smoothing. Payloads would be deployed rather than constructed; related activities would primarily consist of placing, empowering, calibrating, and troubleshooting the payload.

From the global perspective, preliminary analysis indicates that the observatory is best located near the equator on the far side, away from the limb, but not at the far side center. Figure 2.3.6-3 summarizes the major concerns. Equatorial latitudes allow a global view of sky; otherwise, the horizon will permanently conceal regions about the opposite pole. Equatorial latitudes also have more frequent launch windows for rendezvous with equatorial orbits, since the long-period lunar rotation is a relatively minor concern. There is a bit of a trade for an ideal longitude for the observatory. To ensure adequate blockage of terrestrial radiation, the very low frequency array (VLFA) should be located away from the limb. One seeks to maximize the interferometer baseline for the Moon-Earth radio interferometer (MERI). Thus the MERI line of sight should be perpendicular to the Earth-Moon line. This occurs near the zenith at the limb and closer to the horizon at other longitudes. Placement too near the far side center could result in operational constraints since the terrain might mask signals. Longitudes around 45° from the center appear to be a good compromise that also provides a line of sight to a communication satellite either in a halo orbit about the superior libration point or at an equilateral Lagrange point.

From a local perspective, analysis indicates an appreciable latitude in local site selection. Surface roughness is the only local characteristic so far identified, and it does not appear to be a major concern. The terrain should not unduly interfere with launch, landing, and surface transportation. Local terrain requirements for communication and power distribution depend on the specific techniques used, but extreme flatness does not appear to be a major concern. For large arrays, flatness appears not to be so much a concern as precisely knowing the relative placement of array elements. The lunar far side is predominantly highlands and although it appears fairly rough on a global scale, on a local scale the terrain is most likely akin to rolling hills. Thus there appear to be ample opportunities for suitable sites.

The baseline power system is a 50-kWe photovoltaic (PV), with regenerative fuel cells (RFC's) as a power conditioner and power storage device. The PV system will be deployed via roll-out blankets. Once deployed, the blankets will be electrically connected to two 25-kWe RFC. All power needs of the equipment and experiments will be provided by this central power station. Individual cables to connect a piece of equipment to the central supply will be considered part of the equipment setup.

The power systems (50 to 100 kWe) need to be relatively complete and easily deployable. These features are desired in order to keep crew EVA time to a minimum. The large amount of construction time on the Moon projected in this case study can be alleviated by the use of construction devices that are robotic or teleoperated from Earth. Once again, both systems warrant further studies in the upcoming year.

2.3.6.2 Technology Drivers

There are three major technologies needed in this case study: EVA systems, power systems, and teleoperation of construction equipment from Earth. The nature of the work requires an "everyday" EVA system which should provide a higher degree of flexibility than the Apollo vintage suits.

2.3.6.3 Systems Alternatives and Opportunities

Alternative schemes for power systems and construction systems have been suggested and deserve further study for the Lunar Observatory Case Study. For the power systems envisioned in this case study, both nuclear and PV/RFC compete in terms of mass for the 50 kWe user needs. However, there is no demonstrated nuclear system at this size that is also self-deployable. The need for a self-deployable nuclear system arises from the desire to limit the crew EVA time. Further studies on the ease (or difficulty) of constructing nuclear power systems on planetary surfaces are needed to fully resolve this issue.

The need for a central power system may be overemphasized. If indeed the science systems are tens of kilometers apart, the mass of connecting cable and crew time associated with deploying this cable may be so large as to suggest that distributed power systems may be more useful. Ongoing studies will resolve this issue.

Construction on the Moon with devices teleoperated from the Earth provides a challenge to technology but greatly assists the mission planner. Long-term

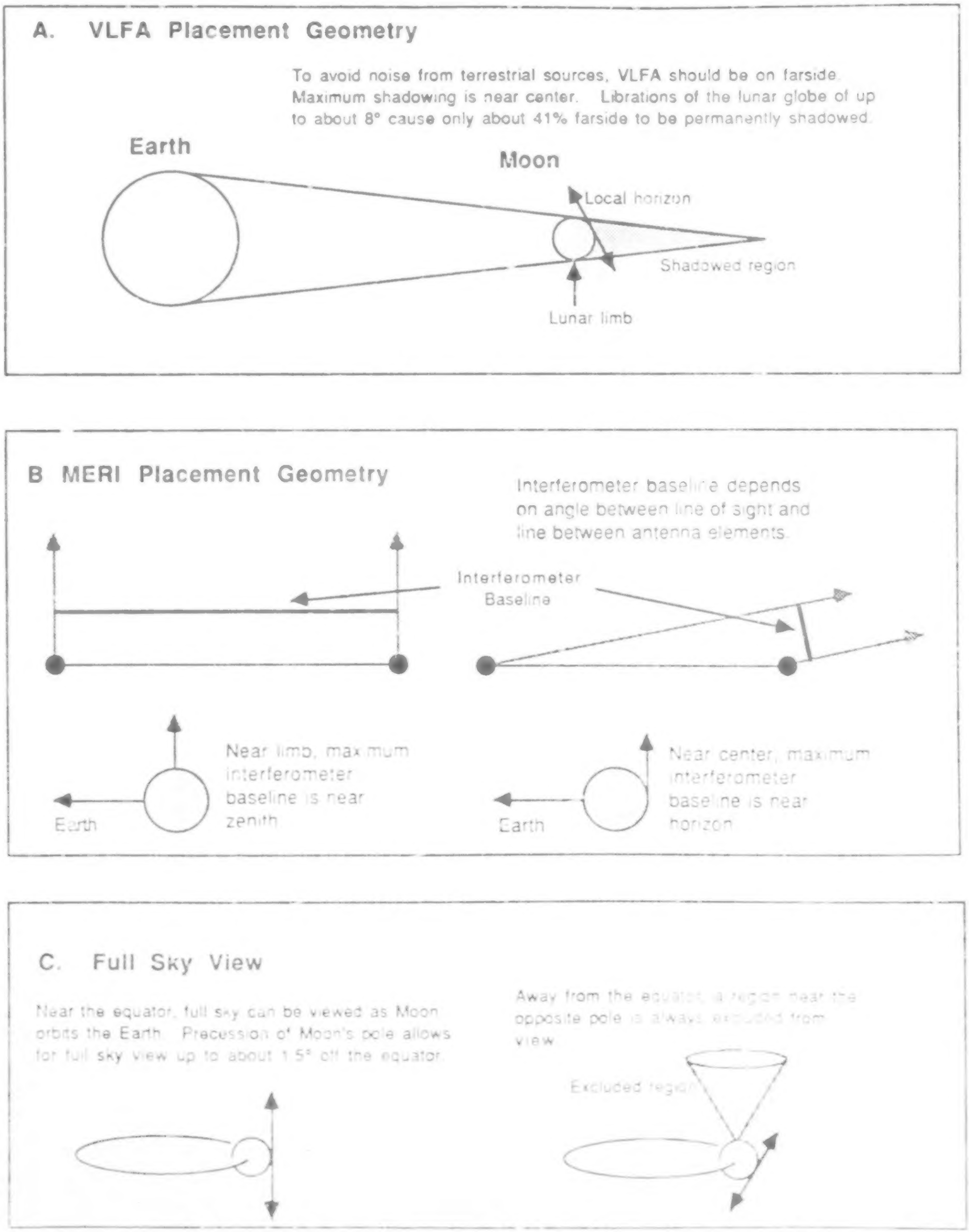
A. VLFA Placement Geometry
To avoid noise from terrestrial sources, VLFA should be on farside. Maximum shadowing is near center. Librations of the lunar globe of up to about 8° cause only about 41% farside to be permanently shadowed.
Earth
Moon
Local horizon
Shadowed region
Lunar limb
B MERI Placement Geometry
Interferometer baseline depends on angle between line of sight and line between antenna elements.
Interferometer Baseline
Near limb, maximum interferometer baseline is near zenith
Earth
Near center, maximum interferometer baseline is near horizon
Earth
C. Full Sky View
Near the equator, full sky can be viewed as Moon orbits the Earth. Precession of Moon's pole allows for full sky view up to about 1.5° off the equator.
Away from the equator, a region near the opposite pole is always excluded from view.
Excluded region

Figure 2.16-3: Lunar observatories placement concerns.

construction could be handled by these devices, or similar robotic devices, in advance of the first crew, provided the 4-second delay in teleoperation between Earth and lunar far side can be mastered (i.e., ≈3 seconds Earth-to-L2 communication satellite; ≈1 second L2-to-lunar far side). Hence, either the surface staytime of the crew could be reduced or more mission objectives could be achieved.

The baseline concept for the observatory layout allows for isolation of the individual elements to alleviate interference and to allow for varying placement requirements. Such requirements are currently not well defined, and, if these concerns are somewhat minor, there will be opportunities for simplifying placement and shortening power distribution and communication links. One option is to deploy the VLFA in a large spiral pattern of separate dipoles and then place the other elements within this spiral. Another option, especially with distributed power supplies, is to deploy elements with substantially different requirements at different locations; e.g., the VLFA near the far side center and the MERI near the limb. Here a major issue is the cost of distributed logistics.

2.3.7 Case Study Synthesis

2.3.7.1 Evaluation of Inputs

Consistency. No major inconsistencies were found in the assumptions used by the IA's or in the results obtained.

Parametric Results. In general, data were not submitted explicitly in parametric form. However, the level of breakdown was generally sufficient to provide a basis for calculation of mass, power, volume, and the like, for alternative cases. In addition to detailed mass breakdowns, the Transportation Agent included data on mass fractions and delta V's, facilitating recalculation.

Options. The definition of Case Study 3 included very few mandatory program options. As a result, the options considered were alternatives that arose during the study. These included reusable rather than expendable vehicles, direct descent to the lunar surface, and a revised program schedule.

Reusable Vehicles. Both the Node and Transportation IA's concluded that reusable vehicles should be considered. The inert weight (and the cost) of the vehicles could be saved each year; although the empty vehicles only represent a relatively small portion of the total mass in LEO, the savings are worthwhile.

Case Study 3 is particularly amenable to reusable LTV's and LEAV's. Both vehicles are returned to the LEO node as an integral part of the mission. Consequently, the design impact of reusability will be minimal. [illegible] pose as many problems as reuse of the vehicles. Except for the two cargo missions during the setup phase, nearly a year is available for the refurbishment required. Aside from other programs outside the scope of Case Study 3 that might utilize the assembly hangar, that facility would be available for storage. The possibility also exists that a reusable LTV could be employed for other missions, such as geosynchronous launches, between lunar visits.

It should be noted that the expendable vs. reusable vehicle question is planned as a trade study in FY 1989 for Case Study 3.

Direct Descent. The Transportation IA also considered a direct lunar descent and direct Earth entry option. In this option, discussed more fully in section 2.3.5.3, the LTV and LDV are combined into a single vehicle that descends directly to the lunar surface without first entering lunar orbit. Trans-Earth injection is performed by the same vehicle directly from the lunar surface. Estimated reduction in mass to LEO is about 14 t for a piloted mission and 29 t for a cargo mission.

Date of First Flight. The ERD schedule was found to be unrealistic. Accordingly, an alternative schedule was developed in which phase A/B for the LEO node was initiated in 1991.

Retaining the ERD-mandated development times resulted in a launch date of 2004 for the first lunar mission.

Special Assessments and Broad Trades. Two special assessments, both concerned with electrical power, were directed specifically toward Case Study 3. The SRD also specified that this case study be included in the node location trade; however, it was found that only a LEO location was reasonable. This trade is discussed more fully in section 4.1.

Power System Selection. The first study compared nuclear and solar PV power systems for the observatory. The principal findings were that (1) the nuclear system required more construction time than the PV and (2) the nuclear system has significantly less mass at the power level estimated for observatory operation (50 kW to [illegible] kW). For the construction phase, where the power system must begin operating as soon as possible, a PV system was considered more feasible.

The Planetary Surface Systems IA found that the mass penalty of the PV system for the operational phase is largely offset by the mass of the extra power [illegible]

Surface Stay Time Extension. The second study concerned extending the crew stay time on the lunar surface by using a PV/fuel cell system to augment the lander's power supply. The primary advantage is the ability to save a launch from Earth by allowing a crew to stay at least 42 days on the surface rather than 14 days. The crew then has the option of finishing delayed tasks or performing more surface activities than originally planned. The net result is a reduction in the total number of missions needed for the operation of the lunar observatory.

These results were not fully utilized by the IA's because the construction time required for the observatory was not defined with sufficient fidelity to warrant missions longer than 14 days. A more detailed assessment of crew time requirements will better use the results of this special study in the upcoming year.

2.3.7.2 Principal Issues and Program Risks

The most important issue identified in this case study is the crew time required for setup of the observatory. It was found that the required work could probably be done. However, the level of confidence in the relevant parameters, and the uncertainty in the definition of the user set to be deployed, leave the results open to question.

EVA systems are a closely related issue. Current technology is oriented primarily toward Shuttle and Space Station Freedom operations, in which onboard maintenance requirements are minimal, mass is not a primary consideration, and the question of dust and similar contamination does not arise. Substantial advances in EVA systems are needed to support the intensive surface activity required by this case study.

Selection of a power system for the lunar observatory requires further study. The current estimates of power required for observatory operation are within a gray area where neither the PV nor the nuclear system has a clear advantage.

The payload mass to be delivered to the lunar surface was derived from an assumed set of astrophysical instruments at the observatory and an assumed set of geophysical instrumentation for the exploration missions. These instrument sets are considered reasonable but lack concurrence by the scientific community. Instrument sets that are generally accepted within the scientific community must be defined before reasonable payload and crew-time requirements can be established. Such definition is also important for the selection of a power system, since both the power level and [illegible] selection.

[illegible]

and payloads at the LEO node, aerocapture systems and techniques, lunar surface EVA systems, performance verification of RL-10 derivative engines, adequate insulation of small cryogenic tanks, onorbit cryogenic fluid transfer, automated rendezvous and docking, and easily deployed power systems suitable for this application. In all cases, however, the challenges were judged to be within the capabilities anticipated for the timeframe under consideration.

2.4 LUNAR OUTPOST TO EARLY MARS EVOLUTION (CASE STUDY 4)

The objective for Case Study 4 is the development of a sustained and evolving human presence beyond low Earth orbit (LEO) for the purpose of enhancing scientific knowledge, facilitating advanced technology development, and encouraging and promoting space enterprise. An evolutionary approach is the central theme for this case study for manned exploration and the establishment of planetary outposts.

2.4.1 Case Study Overview

Results of FY 1989 activities in support of this case study are presented in the following subsections.

2.4.1.1 Key Features

The principal features of this case study is the early emplacement and operation of both a lunar outpost and a Mars outpost (FY 1989 emphasis) followed by the evolution of the outposts into self-sufficient lunar and Mars bases. A key to success in this evolutionary process for self-sustaining bases is the exploration and assessment of lunar and Mars resource potentials and the exploitation of these resources to reduce the need for Earth resupply. Human progression to Mars is accomplished after the establishment of a permanent lunar base.

[illegible]

reducing Earth-to-orbit (ETO) mass transportation requirements. This propellant production capability is extended to the martian moon Phobos to leverage further the reduction of ETO mass and to enhance self-sufficiency. Science instrumentation placed at both the lunar and Mars outpost sites includes the capability for environmental, geophysical, astrophysical, and astronomical data gathering during the long-term course of outpost operations.

Precursor requirements for this case study include acquisition of environmental, topographical, cartographical, and geophysical data at global, regional, and local levels on the Moon, Mars, and Phobos to enable site selections and engineering analyses for the lunar and martian outposts and the Phobos propellant production site. Additionally, a rigorous program of Earth-based, space-based, and lunar-based life sciences research is required and should include variations between zero and one-third g.

2.4.1.2 Mission Profile

Two distinct mission phases are inherent in this case study: the initial buildup of a lunar outpost with parallel technology development followed by the buildup phase for the Mars outpost. Between 2002 and 2004, the first electric cargo vehicle (ECV), its lunar payload including the lunar operations vehicles (LOV's), and fuel are launched and assembled at the LEO node. Following assembly, the ECV begins its outbound spiral and arrives in low lunar orbit (LLO) in 2004. The first lunar surface crew, carried by a conventional lunar piloted vehicle (LPV), also arrives at the Moon. In November 2004, both cargo and crew descend to the lunar surface in separate LOV's (hybrid vehicles which service cargo, crew, and LLOX transport needs). Over the ensuing year the outpost is set up (habitation module, power plant, LLOX plant) and LLOX production is initiated at a rate of 150 t per year. This LLOX production rate is sufficient to support lunar transportation needs. Additional flights support the outpost buildup including crew change out on an annual basis.

In January 2006, the first four-person science crew arrives at the outpost. The biomedical laboratory is operational and the second and third LLOX plants are in production. And by 2007 the outpost facilities have expanded to a full operational status including the introduction of the fourth LLOX plant which brings the LLOX production to a rate of 600 t per year (the rate necessary for the added Mars mission phase support).

With the required technical, operational, and scientific capabilities in place, the second mission phase begins — Mars outpost buildup. Similar to Case Studies 1 and 2, a split mission concept is employed with the interplanetary payload split between a cargo carrier and crew carriers. Sprint class trajectories for the manned flights were not used for Case Study 4, however, because of the 1 to 2 year planned staytimes at Mars. Rather, near-fuel-minimum transfers were employed. These are characterized by an opposition-class outbound leg which provides a flyby abort capability coupled with a conjunction-class return leg.

Between 2007 and early 2010, an ECV, its Mars payload, and fuel are launched and assembled at the LEO node. Following assembly in May 2010, the ECV begins its outbound spiral from Earth for its direct journey to Mars. The cargo vehicle arrives in high circular orbit about Mars in November 2012 (see figure 2.4.1-1). The three Mars logistics landers (MLL's) detach from the cargo vehicle and begin their descent to the Mars surface. The cargo vehicle continues with its downward spiral to a subsequent rendezvous with the martian moon Deimos. Upon arrival at Deimos, two robotic payloads are detached for surface exploration. Continuing with the downward spiral, the cargo vehicle releases three communications satellites at synchronous altitude above Mars. Finally, the cargo vehicle arrives at the martian moon Phobos, deploys the water and propellant production facility on the Phobos surface, and awaits the arrival of the first Mars crew with its remaining payload of four Mars crew sortie vehicles (MCSV's) which support crew descent to the Mars surface, ascent, and orbital operations.

Beginning in 2010, the Mars piloted vehicle (MPV) is launched, assembled, fueled with liquid hydrogen, and mated to an ECV at the LEO node. Following assembly and mating, the ECV begins an outbound spiral and transports the vehicle to low lunar orbit for fueling with LLOX (see figure 2.4.1-2). The ECV then transports its cargo to a high circular orbit about Earth, detaches, and begins a spiral back to LEO. The Mars crew departs from LEO in an LPV to rendezvous with and man the MPV. Following final vehicle checkout, the crew departs Earth in November 2013, on a fuel-minimum outbound trajectory for Mars (see figure 2.4.1-3). Upon arrival in August 2014, the manned vehicle aerobrakes in the martian atmosphere with a subsequent rendezvous at Phobos. The crew monitors the Phobos propellant operation and then mans an MCSV and descends to the Mars surface in the vicinity of the awaiting logistics landers. Thus begins the Mars outpost buildup. Initial activities include the gathering of surface data required to make final equipment and site preparation for the Mars habitat. Following site preparation the habitat is set up and other activities proceed, including unpressurized and pressurized rover Mars surface traverses. Following their nominal staytime, the crew secures the outpost, mans the crew sortie vehicle, and ascends for rendezvous with the MPV at Phobos. The crew transfers to the MPV and departs Mars on a fuel-minimum transfer in September 2015, with Earth arrival in August 2016.

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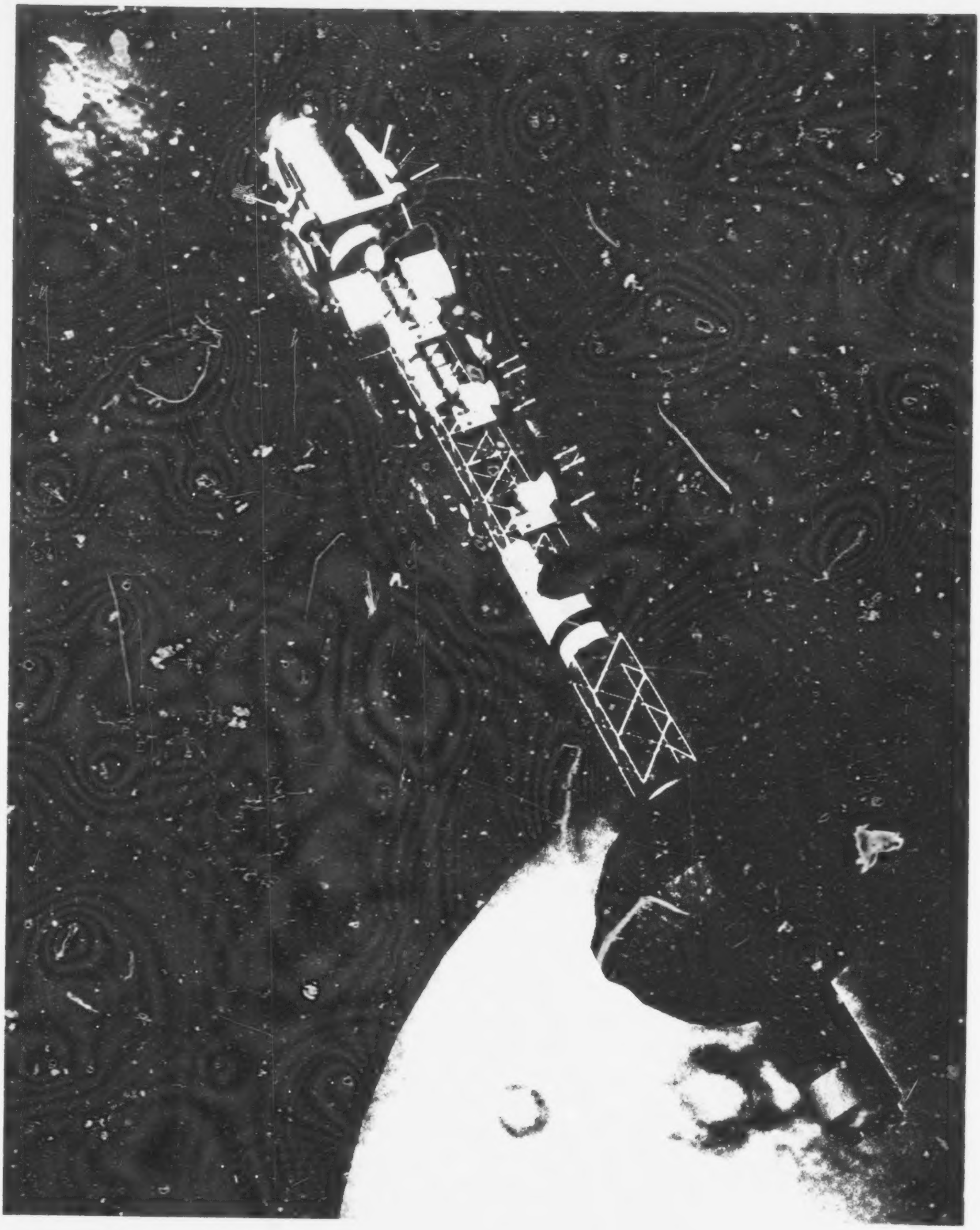

Figure 2.4.1-1.- The electric cargo vehicle arrives in the Mars system on its way to Phobos.

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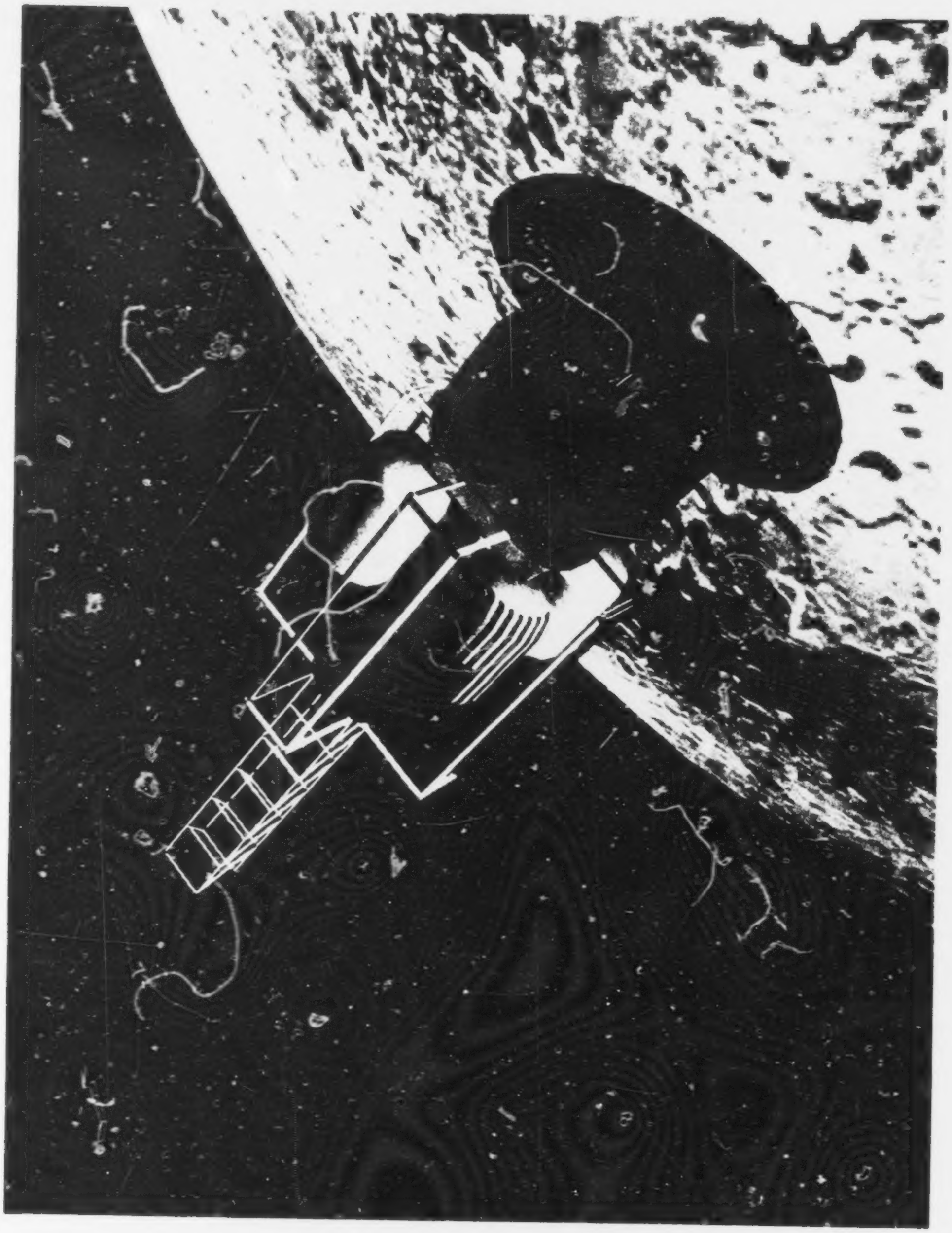

Figure 2.4.1-2.- The electric cargo vehicle arrives in low lunar orbit for fueling of the Mars piloted vehicle.

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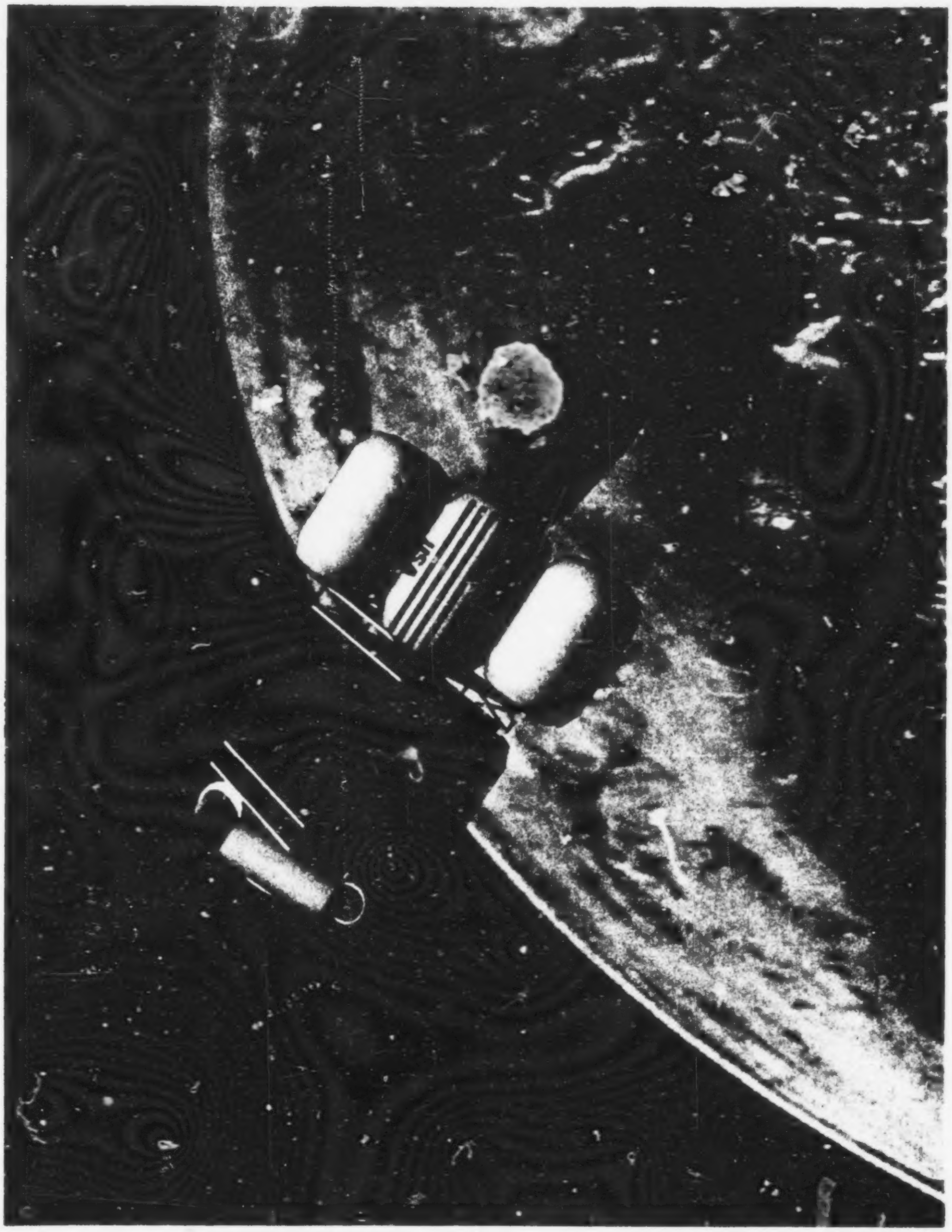

Figure 2.4.1-3.- The Mars piloted vehicle departs Earth following the trans-Mars injection maneuver.

Two subsequent manned flights on successive Earth-Mars opportunities bring the Mars outpost to a full operational status. The second crew arrives at Mars in July 2016, and departs Mars in March 2018, with an Earth arrival in October 2018. The third crew arrives at Mars in March 2018, and departs Mars in June 2020, with an Earth arrival in December 2020. For the piloted flights for this case study, the trajectory strategies employed for outbound transfer to Mars (and propellant loading) permit both a flyby abort capability and an opportunity for an Earth return up to 60 days after arrival at Mars. The nominal Earth return opportunity is a near-fuel-minimum transfer. Extended staytime at Mars beyond the nominal can be accommodated using Phobos-produced propellants. The staytime options at Mars are illustrated in figure 2.4.1-4.

The operational lunar and Mars outposts, along with the Phobos water and propellant production facility, provide an extensive "multiplanet" infrastructure. This infrastructure will be the stepping-stone for self-sustaining bases as the overall exploration knowledge base expands and the Earth-resupply umbilical becomes progressively severed.

2.4.1.3 Summary Data

Mission vehicles and payloads are assembled, checked out, and serviced at the LEO node. The LEO node stores propellants (up to 366 t) delivered by the heavy-lift launch vehicle (HLLV) and transfers them to the mission vehicles. It also houses mission crews of eight before departure to and after return from the Moon and Mars, along with assembly and checkout crews of three. A life sciences support facility is provided at the node for three crewmembers.

The total mass through LEO increases to a steady-state value of approximately 318 t per year in support of the emplacement of lunar and Mars outposts. During the buildup phase for the lunar outpost, 316 t of support equipment and user payloads are delivered to the lunar surface. During the Mars outpost buildup, 86 t are delivered to Phobos for the propellant production facility and 135 t are delivered to the Mars surface.

Assumed User Set. For purposes of analysis, the following capabilities were assumed to be included for science support:

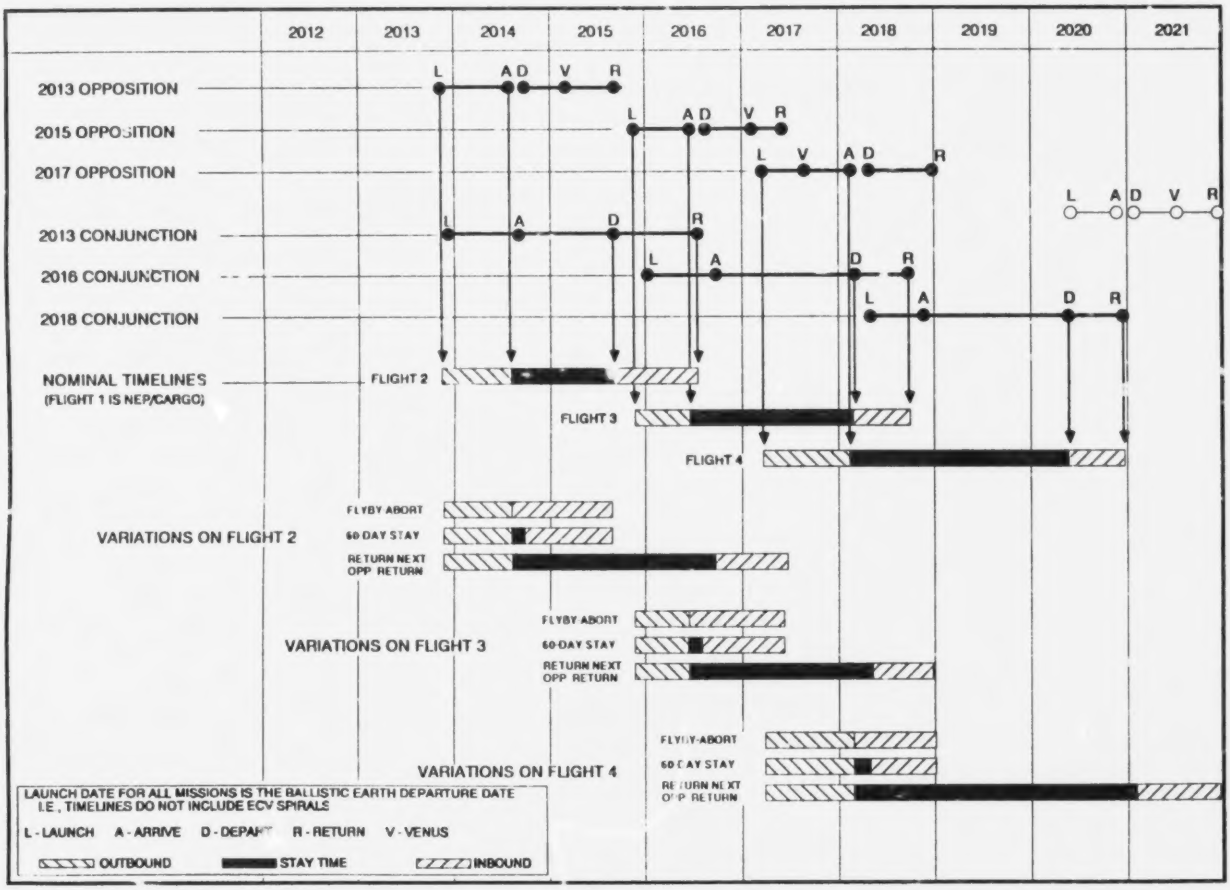

Figure 2.4.1-4.- Crew staytime options for piloted Mars flights.

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a. Lunar science/astronomy
   (1) 3.3 t of mapping, imaging, and monitoring orbiters
   (2) 39 t of teleoperated unmanned exploration equipment
   (3) Four-man science-dedicated crews
   (4) 54 t of lunar science and astronomical equipment
   (5) Global manned site exploration capability
   (6) 70 t of LLOX/year for global manned exploration landers

b. Lunar life sciences — 19 t of life sciences equipment

c. Mars science
   (1) 58 t of Mars surface science equipment
   (2) 10 t Deimos robotic exploration capability

Additionally, rover-supported extravehicular activity (EVA) capability is provided as follows:

a. Lunar — Local EVA with unpressurized rover (10 km)

b. Mars — Local and regional EVA's with unpressurized and pressurized rovers, respectively (10 km and 100 km)

2.4.2 Mission Definition and Manifest

The Earth-to-LEO manifest for Case Study 4 is presented in table 2.4.2-I. The dates indicate the year in which the payload is delivered to LEO. Delivery dates to the payload's final destination vary significantly due to the long flight times required for ECV spirals and Mars

TABLE 2.4.2-I.- CASE STUDY 4, EARTH TO LEO MANIFEST

	Lunar Outpost to Early Mars Evolution - Payload Element Manifest																						
MSDB NUMBER	SURFACE SYSTEMS SCIENCE ELEMENT NAME	MASS (KG)	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16
LOM 001-A	Imaging Lunar Polar Orbiter	600		1																			
LOM 005-A	Cartographic Lunar Polar Orbiter	300			1																		
LOM 002-A	Lunar Polar Orbiter	1200	1																				
LOM 003-C	Lunar Monitoring Orbiter	300			2	2																	
LSM 001-C	Geophysical Station Network Mission	1210					4	4		4	4												
LSM 006-C	Unmanned Sample Collection Mission	2110				2		2		2	2	2	2	2	2	2	2	2	2	2	2	2	2
LSM 008	Local Unmanned Traverse Mission	2865			1																		
LSE 009	Unpressurized, 10 km Manned Rover	560								1													
None	Manned Site Expedition Module	6000										1											
None	Bio-Medical Laboratory	19000									1												
None	TBD Lunar Science Equipment	54											1										
None	Lunar Supplies & Resource Equipment	Varies									X	X				X		X		X			X
MSDB NUMBER	SURFACE SYSTEMS SUPPORT ELEMENT NAME	MASS (KG)	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16
	Lunar Portion:																						
LOE 001-A	L2 Halo Comsat	2200	1	1																			
NA	Power Unit	6000								1													
NA	Digger/Crane	3600								1													
NA	LLOX Plant	40000								1		3											
NA	Habitation Module, Air Locks, etc.	21500								1													
NA	Lunar Lander Facilities	8000								1													
NA	Bagger	250								1													
NA	Excavator Truck	1900								1													
NA	Teleoperated Assistant	800								1													
	Mars Portion:																						
NA	Mars Comsat	4000												3									
NA	Deimos Teleoperated Vehicle	5000												2									
NA	Phobos Propellant Plant	86000													1								
NA	TBD Mars Payload	45000												3									
	Lunar Outpost to Early Mars Evolution - Payload Element Manifest																						
MSDB NUMBER	TRANSPORTATION SYSTEMS ELEMENT NAME	MASS (KG)	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16
	Lunar Portion:																						
NA	Expendable Launch Vehicle	TBD	2	2	4	4	4	6		6	6	2	2	2	2	2	2	2	2	2	2	2	2
NA	Lunar Piloted Vehicle	7900									1		1										
NA	Lunar Personnel Lander	7500									1	1											
NA	Lunar Cargo Lander	5200									2												
NA	Lunar Propellant Tanker	14600										2											
NA	Electric Cargo Vehicle	125000							1		1							1					
	Mars Portion:																						
NA	Mars Logistics Lander	10000												3									
NA	Mars Crew Sortie Vehicle	20000												4									
NA	Mars Piloted Vehicle	110000															1		1		1		
NA	Outbound Propulsion Stage	63000															1	1		1			
NA	Return Propulsion Stage	15000													1			1		1			
	Total Propellant*:																						
NA	ECV Fuel - Argon	10000								8		7	14			14	7		13		13	7	13
NA	Liquid Hydrogen	1000									43	36	65	5	5	5	46	66	27	109	18	81	18
NA	Liquid Oxygen	1000									69	44	33	33	33	33	33	60	164	50	33	50	33
* Note: The propellant presented in this section represents the total of each fuel/oxidizer brought to LEO during that year.																							
The propellant is not necessarily used the year it's brought to LEO. Further, the totals do not represent the total propellant in the system.																							
Propellant is produced on the Moon and on Phobos and integrated into the system. This is only the Earth to LEO manifest.																							
MSDB NUMBER	NODE SYSTEMS ELEMENT NAME	MASS (KG)	96	97	98	99	00	01	02	03	04	05	06	07	08	09	10	11	12	13	14	15	16
NA	LEO Vehicle Assembly Facility	112000								1													
NA	LEO Propellant Storage Facility	45000						1															

mission trip durations. Figure 2.4.2-1 illustrates the timing of the flights leaving LEO for the lunar portion of the case study. The timing of flights leaving LEO for the Mars phase is shown in figure 2.4.2-2. Manifested payloads on the Mars cargo flight are shown in table 2.4.2-II. Manifested payloads on the piloted flights are shown in table 2.4.2-III.

2.4.3 Mission Architecture and Infrastructure

The mission architecture for the lunar phase of Case Study 4 is illustrated in figure 2.4.3-1 and in figures 2.4.3-2 to 2.4.3-3 for the Mars phase of the case study. As shown in these figures, HLLV's provide the Earth-to-orbit transportation. The Shuttle provides transportation for the assembly and mission crews to and from the LEO node.

Mass delivery and crew requirements for the lunar phase are summarized in figure 2.4.3-4. Similarly, requirements for the Mars phase are shown in figure 2.4.3-5 with integrated summary requirements for the case study in figure 2.4.3-6.

Major programmatic milestones are shown in figure 2.4.3-7. The milestones shown reflect the requirement for serial phase C/D schedules for the lunar and Mars phases of the case study. Prerequisite milestones are shown in figure 2.4.3-8.

2.4.4 Transportation Systems Definition

The results and descriptions presented in this section for Case Study 4 include no inputs from the transportation integration agent (IA). Instead, only the initial Mission Analysis and Systems Engineering (MASE) group-derived inputs are discussed.

Case Study 4 requires the development of a number of transportation capabilities to meet the variety of mission objectives and constraints. The different environments under which the transportation systems are to operate create a situation in which vehicle synergism is difficult. Additionally, the mass-to-LEO constraints of the case study and the long duration of this aggressive space program allow little room for multipurpose transportation systems. Such systems typically sacrifice efficiency in a specific mission for general applicability to a variety of missions. However, these concerns must be balanced by the realization that developing a score of specialized vehicles is not feasible. Thus, Case Study 4 concentrates on using common vehicle systems and subsystems where possible. For example, the three vehicles traveling from the lunar surface to low lunar orbit and back (LS-LLO-LS vehicles) use a common core system, the lunar operations vehicle, adding on personnel modules or propellant tanks when required.

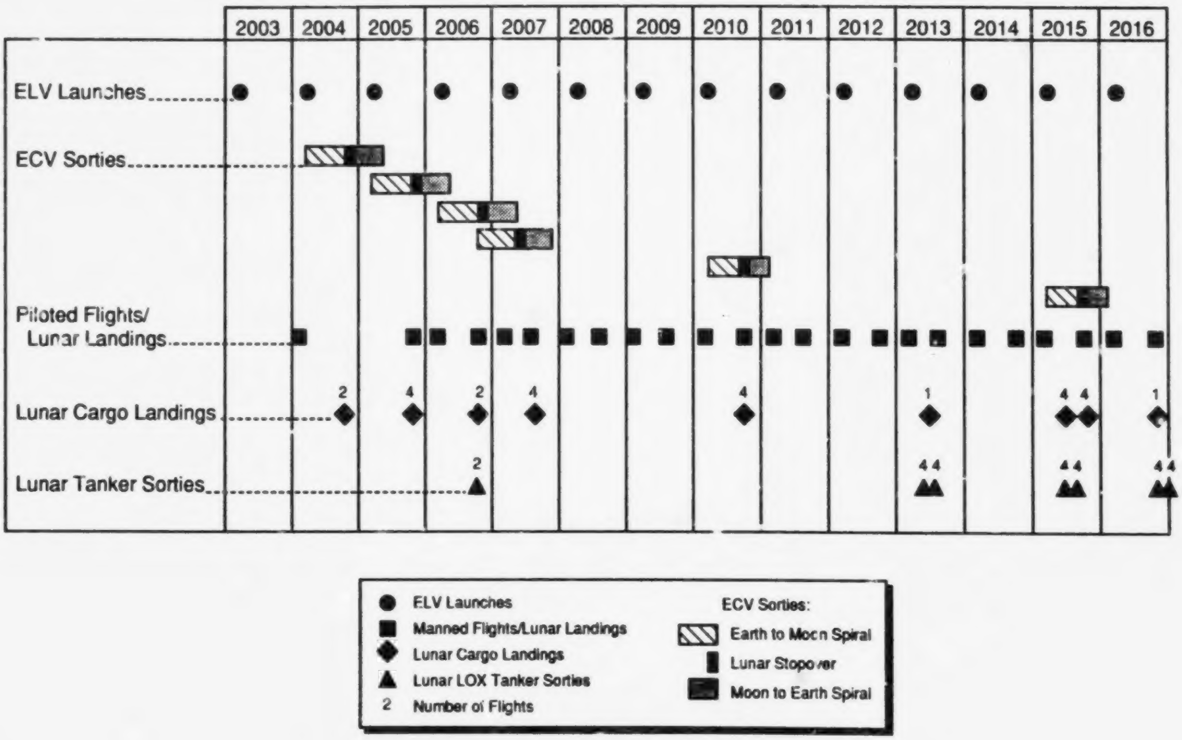

Figure 2.4.2-1.- Case Study 4, lunar portion schedule.

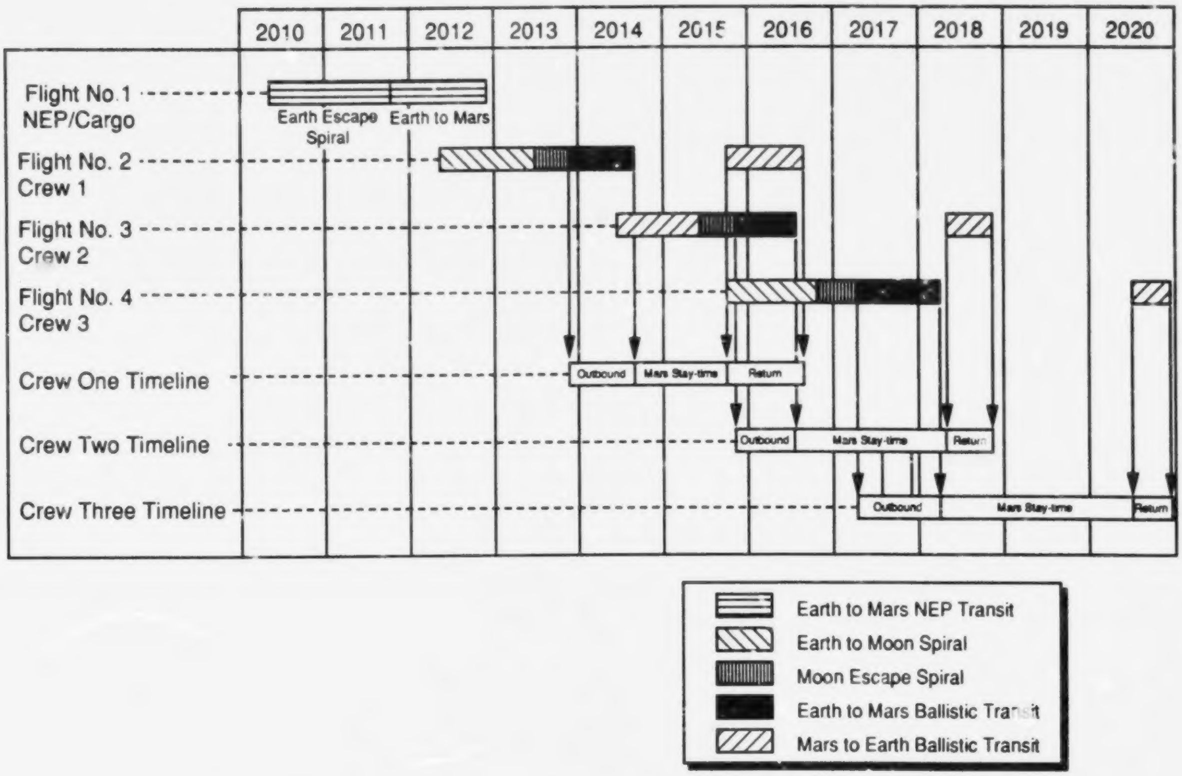

Figure 2.4.2-2.- Mars mission schedule.

TABLE 2.4.2-II.- MANIFEST FOR MARS CARGO FLIGHT

MSDB NO.	ELEMENT NAME	MASS (t)	2010	11	12	13	14	15
	CARGO FLIGHT							
TBD	Mars Logistic Lander (Dry)	55.0	3					
	- LH$_2$ Propellant for MLL	2.5	3					
	- LLOX Propellant for MLL	17.5	3					
	Mars Crew Sortie Vehicle (Dry)	20.0	4					
	- LH$_2$ Propellant for MPEM	7.5	2					
	- LLOX Propellant for MPEM	52.5	2					
	Deimos Robotic Explorers	5.0	2					
	Mars Comsat	4.0	3					
	Phobos Propellant Production Plant	30.0	2					
	Argon Propellant	186.0	1					

TABLE 2.4.2-III.- MANIFEST FOR MARS PILOTED FLIGHT

MSDB NO.	ELEMENT NAME	MASS (t)	2010	11	12	13	14	15
	PILOTED FLIGHT							
TBD	I/P CREW HABITAT	110.0			1		1	1
	- Outbound Propulsion System (Dry)	63.0			1		1	1
	- LH_2 Propellant for outbound flight	52.0			1		1	1
	- LLOX Propellant for outbound flight	368.0			1		1	1
	LH_2 Tanks for LTL sorties	7.0			1		1	1
	LH_2 Propellant for LTL sorties	50.0			1		1	1
	Argon Propellant	122.0			1		1	1
	Lunar Base Cargo	40.0			1		1	1
	Return Propulsion System(Dry)	15.0			1		1	1
	LH_2 Propellant for Crew Return	13.0			1		1	1

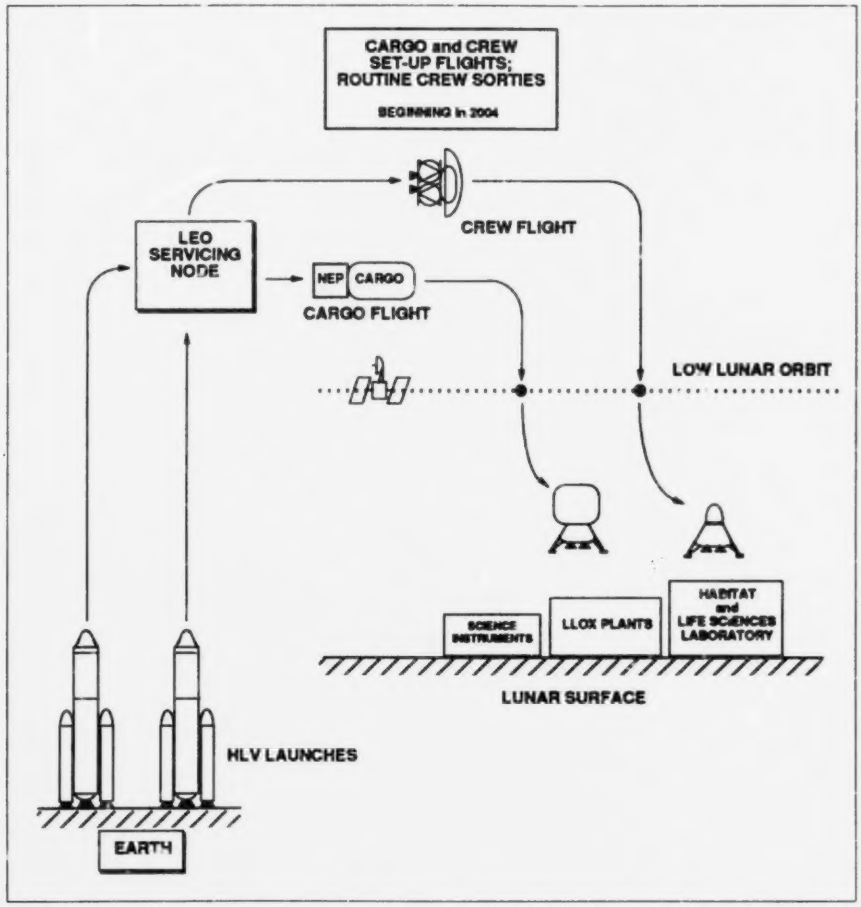
CARGO and CREW
SET-UP FLIGHTS;
ROUTINE CREW SORTIES
BEGINNING in 2004
CREW FLIGHT
LEO SERVICING NODE
NEP
CARGO
CARGO FLIGHT
LOW LUNAR ORBIT
SCIENCE INSTRUMENTS
LLOX PLANTS
HABITAT and LIFE SCIENCES LABORATORY
LUNAR SURFACE
HLV LAUNCHES
EARTH

Figure 2.4.3-1.- Mission architecture for lunar outpost buildup.

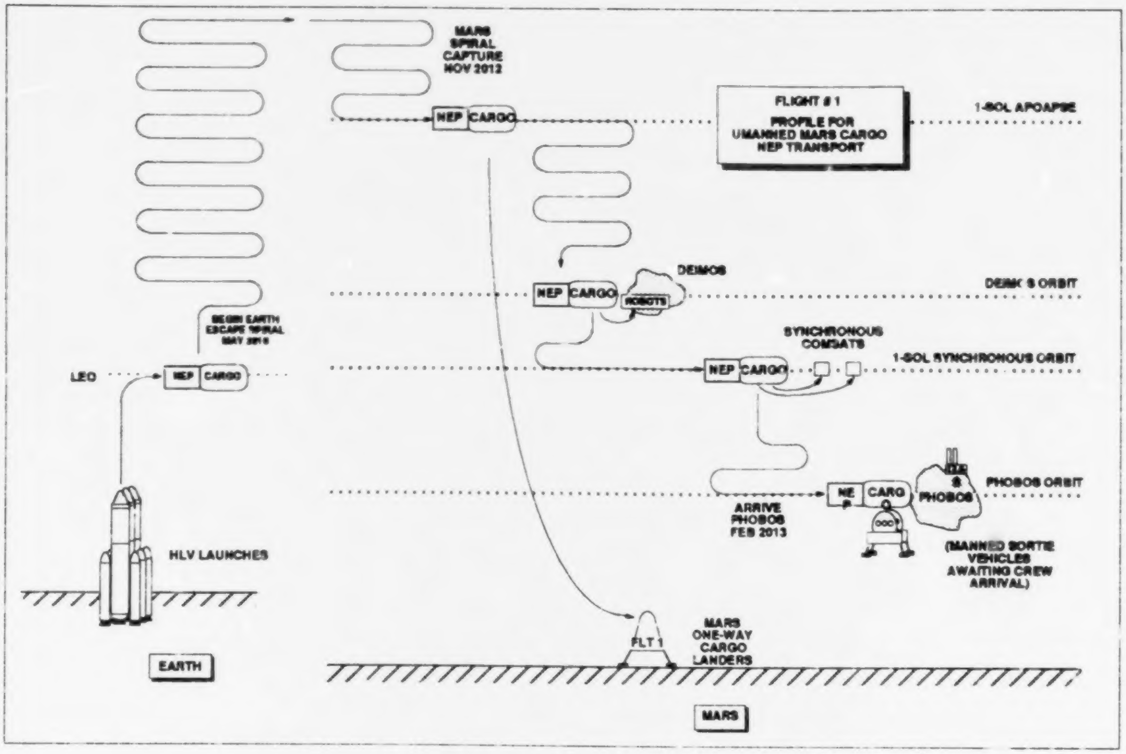
MARS SPIRAL CAPTURE NOV 2012
NEP
CARGO
FLIGHT # 1
PROFILE FOR UMANNED MARS CARGO NEP TRANSPORT
1-SOL APOAPSE
DEIMOS
DEIMOS ORBIT
ROBOTS
BEGIN EARTH ESCAPE SPIRAL MAY 2010
SYNCHRONOUS COMSATS
1-SOL SYNCHRONOUS ORBIT
LEO
PHOBOS ORBIT
NE P
CARG
PHOBOS
ARRIVE PHOBOS FEB 2013
(MANNED SORTIE VEHICLES AWAITING CREW ARRIVAL)
HLV LAUNCHES
FLT 1
MARS ONE-WAY CARGO LANDERS
EARTH
MARS

Figure 2.4.3-2.- Mission architecture for Mars cargo flight.

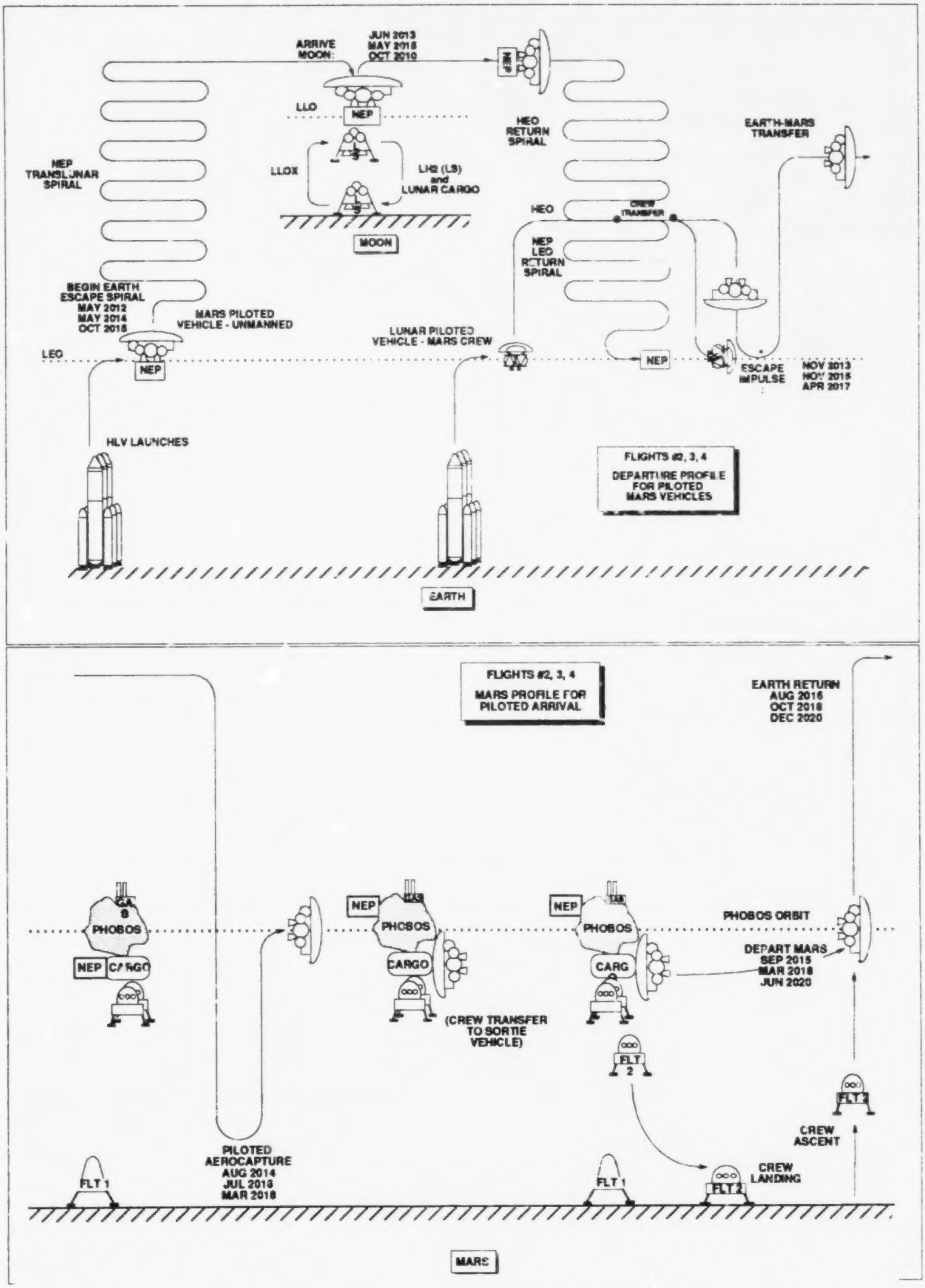
ARRIVE MOON:
JUN 2013
MAY 2016
OCT 2010
NEP
LLO
NEP
LLOX
LH2 (LS) and LUNAR CARGO
MOON
NEP TRANSLUNAR SPIRAL
BEGIN EARTH ESCAPE SPIRAL
MAY 2012
MAY 2014
OCT 2015
MARS PILOTED VEHICLE - UNMANNED
LEO
NEP
HEO RETURN SPIRAL
EARTH-MARS TRANSFER
HEO
CREW TRANSFER
NEP LEO RETURN SPIRAL
LUNAR PILOTED VEHICLE - MARS CREW
NEP
ESCAPE IMPULSE
NOV 2013
NOV 2015
APR 2017
HLV LAUNCHES
FLIGHTS #2, 3, 4
DEPARTURE PROFILE FOR PILOTED MARS VEHICLES
EARTH
FLIGHTS #2, 3, 4
MARS PROFILE FOR PILOTED ARRIVAL
EARTH RETURN
AUG 2016
OCT 2018
DEC 2020
PHOBOS
NEP
CARGO
NEP
PHOBOS
CARGO
NEP
PHOBOS
CARG
PHOBOS ORBIT
DEPART MARS
SEP 2015
MAR 2018
JUN 2020
(CREW TRANSFER TO SORTIE VEHICLE)
FLT 2
FLT 2
CREW ASCENT
PILOTED AEROCAPTURE
AUG 2014
JUL 2016
MAR 2018
FLT 1
FLT 1
FLT 2
CREW LANDING
MARS

Figure 2.4.3-3.- Mission architecture for Mars piloted flights.

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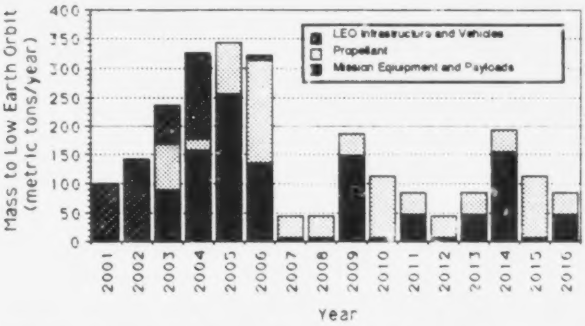
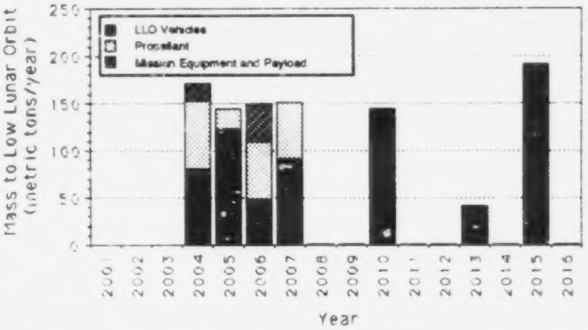
Lunar to Mars Evolution



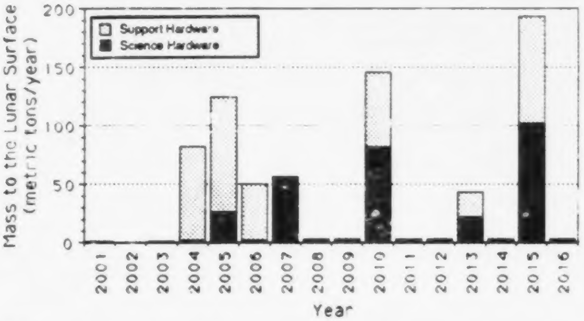
Note: Mass to Low Earth Orbit

LEO Infrastructure and Vehicles includes the vehicle assembly/propellant depot and the dry lunar piloted vehicles (LPV's) and the dry electric cargo vehicles (ECV's), minus mission equipment and payloads
Propellant includes the propellant delivered from Earth and used by the LPV's (LOX/LH2) and the ECV's (argon) in traveling from LEO to LLO and then back
Mission Equipment and Payloads includes all support and science hardware, LLO vehicles and their propellant (LOX/LH2), and the consumables used in traveling between Earth and the Moon



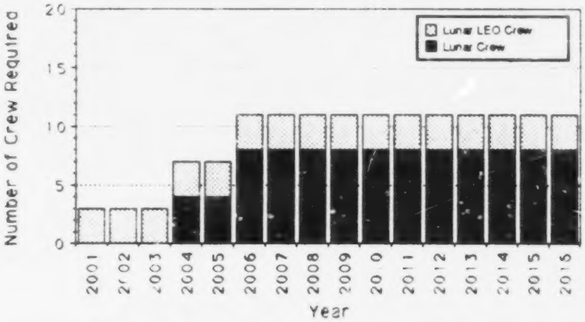
Note: Mass to Low Lunar Orbit

LLO Vehicles includes the dry lunar personnel landers (LPL's), the dry lunar propellant tankers (LPT's), and the dry lunar cargo landers (LCL's)
Propellant includes the propellant brought from Earth used in traveling between LLO and the lunar surface
Mission Equipment and Payload includes the support and science hardware to be taken to the lunar surface, the consumables to be used on the on the lunar surface, and other hardware to be deployed from LLO (e.g., teleoperated landers, lunar satellites)



Note: Mass to Lunar Surface

Support Hardware includes the equipment used to support the lunar outpost (e.g., space suits, habitation modules LLOX plants)
Science Hardware includes the equipment used to conduct the science experiments on the Moon (e.g., geophysical stations, geological exploration equipment)



Note: Number of Crew Required

Lunar LEO Crew is the crew used to construct the assembly/propellant deepot and to integrate and prepare payloads and OTV's bound for the Moon
Lunar Crew is the crew sent to the lunar surface
Note: Lunar LEO crew tour of duty = 180 days
Lunar crew tour of duty = 360 days

Figure 2.4.3-4.- Lunar Outpost to Early Mars Evolution--Lunar portion transportation requirements.

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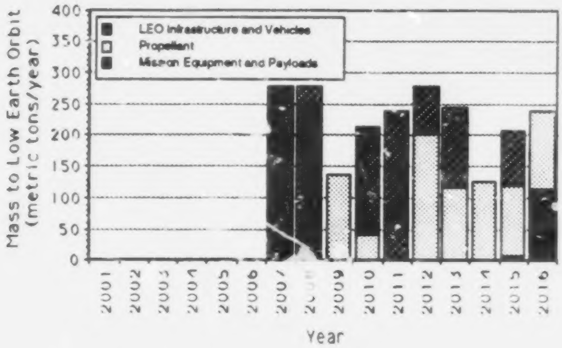
Lunar to Mars Evolution

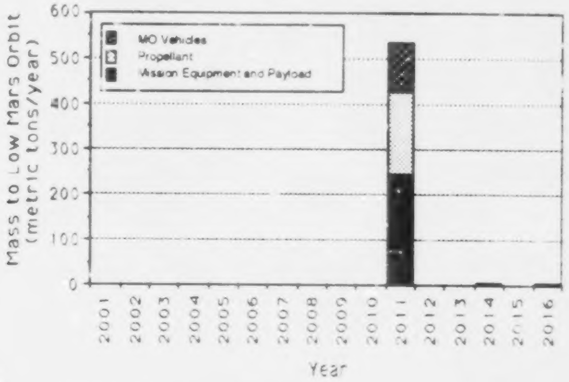


Note: Mass to Low Earth Orbit

LEO Infrastructure and Vehicles includes additional vehicle assembly/propellant storage facilities, the dry Mars piloted vehicles (MPV's) (minus mission equipment and payload), and the dry ECV's used to ferry the dry MPV's to LLO for fueling and to deliver cargo to Mars

Propellant includes the propellant used by the ECV (argon) in traveling from Earth to Mars, the propellant used by the ECV's (argon) in traveling from Earth to LLO (to ferry the dry MPV to LLO for fueling) and then back to Earth, the hydrogen used by the lunar propellant tanker in delivering LLOX to LLO for use in the MPV's, the hydrogen used in the MPV's, and the propellant used by the LPV (LOX/LH2) in delivering the Mars crew to the MPV prior to the Mars departure burn

Mission Equipment and Payloads includes all support and science hardware, MO vehicles and their propellant (LOX/LH2), and the consumables used in traveling between Earth and Mars

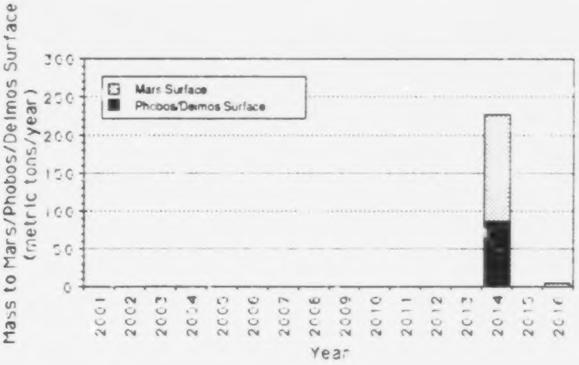


Note: Mass to Mars Orbit

MO Vehicles includes the dry Mars crew sortie vehicles and the dry Mars logistics landers

Propellant includes the propellant brought from Earth (LOX/LH2) used in traveling between Mars orbit and the surfaces of Mars, Phobos, and Deimos

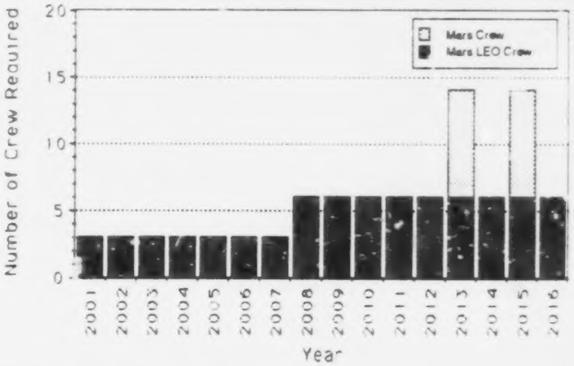
Mission Equipment and Payload includes the support and science hardware brought to the surfaces of Mars, Phobos, and Deimos; the consumables used on the sorties to the surfaces; and other hardware deployed from Mars orbit (e.g., teleoperated landers,Mars satellites, etc.)



Note: Mass to Mars/Phobos/ Deimos Surfaces

Phobos/Deimos Surface includes the equipment used to support and conduct the science experiments on Phobos and Deimos (e.g., space suits, MMU's, simple seismic networks, sample collection sets), as well as the propellant production plant

Mars Surface includes the equipment used to support and conduct the science experiments on Mars (e.g ,space suits, radiation protection garments, meteorological balloons, geophysical/meteorological stations, etc), and to establish an outpost on Mars (e.g., habitation modules, airlocks, supplies)



Note: Number of Crew Required

Mars Crew is the eight-person crew sent to Mars

Mars LEO Crew is the crew used to operate an assembly/propellant depot and construct the trans-Mars vehicles

Mars crew tour of duty = 3 to 4 years

Mars LEO support crew tour of duty = 180 days

Figure 2.4.3-5.- Lunar Outpost to Early Mars Evolution — Mars portion transportation requirements.

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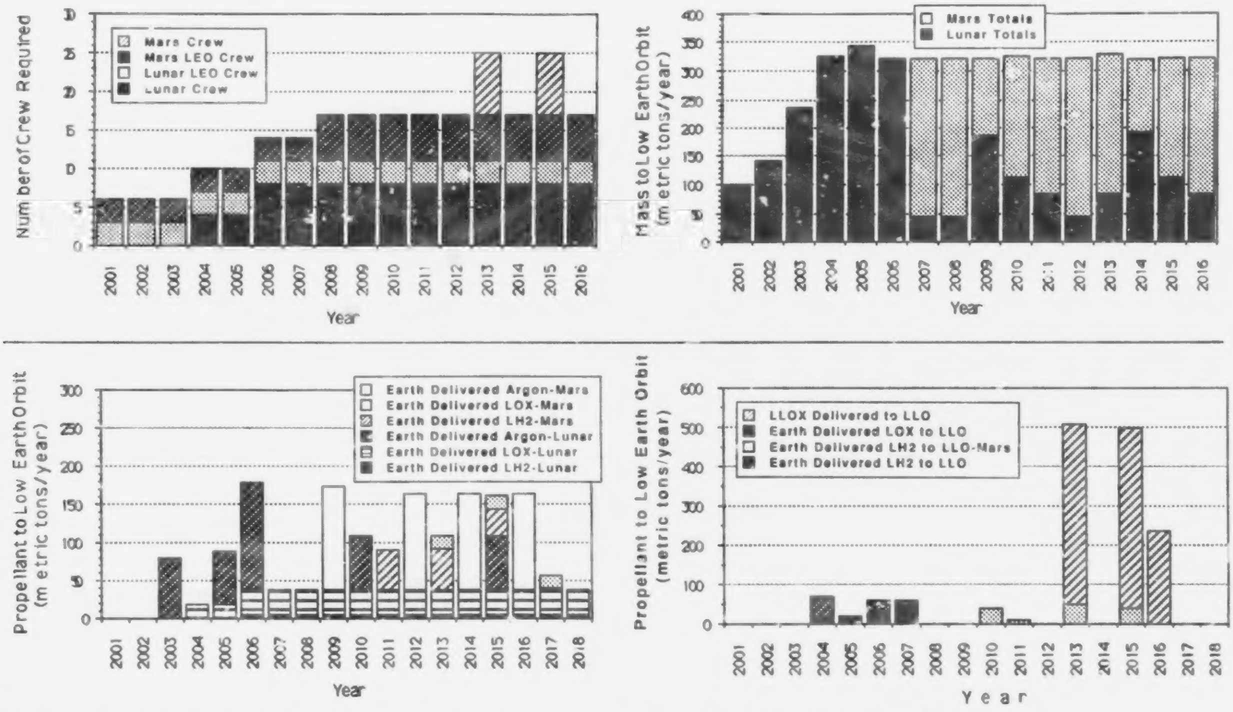
Lunar to Mars Evolution



Propellant to Low Earth Orbit

Earth Delivered Argon-Mars is the propellant (argon) delivered to LEO for use in the Electric Cargo Vehicles (ECVs) for ferrying the Mars Piloted Vehicles (MPVs) to LLO for fueling with LLOX, and for use in the ECV used to deliver the first cargo to Mars.

Earth Delivered LOX-Mars is the liquid oxygen delivered to LEO for use in the Lunar Piloted Vehicle (LPV) when the LPV is used to deliver the Mars crew to the MPV prior to the MPV's Earth departure burn.

Earth Delivered LH2-Mars is the liquid hydrogen delivered to LEO for use in the MPV and the Lunar Piloted Vehicle (LPV) when the LPV is used to deliver the Mars crew to the MPV prior to the MPV's Earth departure burn.

Earth Delivered Argon-Lunar is the propellant (argon) delivered to LEO for use in the ECVs for ferrying the Lunar Outpost cargo to LLO.

Earth Delivered LOX-Lunar is the liquid oxygen delivered to LEO for use in the Lunar Piloted Vehicle (LPV) when the LPV is used to deliver the Lunar crew to LLO and return them to LEO.

Earth Delivered LH2-Lunar is the liquid hydrogen delivered to LEO for use in the MPV and the Lunar Piloted Vehicle (LPV) when the LPV is used to deliver the Lunar crew to LLO and return them to LEO.

Propellant to Low Lunar Orbit

LLOX Delivered to LLO is the Lunar produced liquid oxygen to LLO for use in the MPVs.

Earth Delivered LOX to LLO is the Earth produced liquid oxygen delivered to LLO for use in the Lunar Personnel Lander (LPL), the Lunar Cargo Lander (LCL), and the Lunar Propellant Tanker (LPT).

Earth Delivered LH2 to LLO-Mars is the Earth produced liquid hydrogen delivered to LLO for use in the LPT when the LPT is used to deliver LLOX to LLO for the MPVs.

Earth Delivered LH2 to LLO is the Earth produced liquid hydrogen delivered to LLO for use in the LPL, the LCL, and

Figure 2.4.3-6.- Lunar Outpost to Early Mars Evolution -- integrated transportation requirements.

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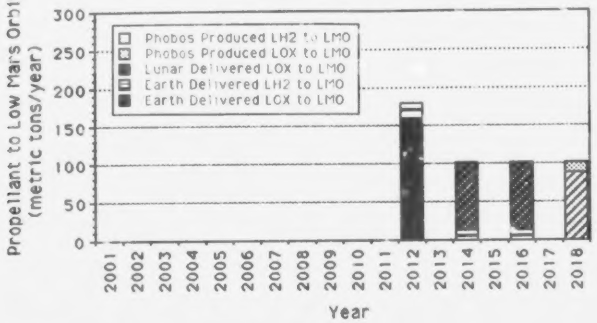


Propellant to Low Mars Orbit

Phobos Produced LH2 to LMO is the liquid hydrogen produced on Phobos for use in the trans-Earth injection burn of the MPV.
Phobos Produced LOX to LMO is the liquid oxygen produced on Phobos for use in the trans-Earth injection burn of the MPV.
Lunar Delivered LOX to LMO is the liquid oxygen produced on the Moon for use in the trans-Earth injection burn of the MPV.
Earth Delivered LH2 to LMO is the liquid hydrogen produced on Earth for use in the trans-Earth injection burn of the MPV and for use in the MLLs and the MCSVs.
Earth Delivered LOX to LMO is the liquid oxygen produced on Earth for use in the MLLs and the MCSVs.

Figure 2.4.3-6.- Concluded.

All vehicles in Case Study 4 are reusable by definition. Maintenance and refurbishment of the vehicles occurs in LEO, on the surface of the Moon, Mars, or Phobos, or on the ECV.

Electric Cargo Vehicle. The ECV is the primary cargo-carrying element of both the lunar and Mars portions of Case Study 4. This vehicle employs a nuclear reactor to power ion engines, producing a constant, low-thrust means of propulsion. Three are used in the case study, one of which delivers the first load of cargo to Mars and remains at Phobos. The first ECV is delivered to LEO in 2002. The first operational flight occurs in 2004 with the commencement of the LEO to LLO spiral and the eventual delivery of the first lunar cargo to the lunar surface. This ECV is used for one additional lunar cargo mission before being used for the first Mars cargo mission in 2010.

The issue of humans in the presence of nuclear reactors for the ECV has yet to be resolved. Consequently, future studies are needed to resolve this issue, and the power system and the performance parameters presented should be considered very preliminary for this vehicle.

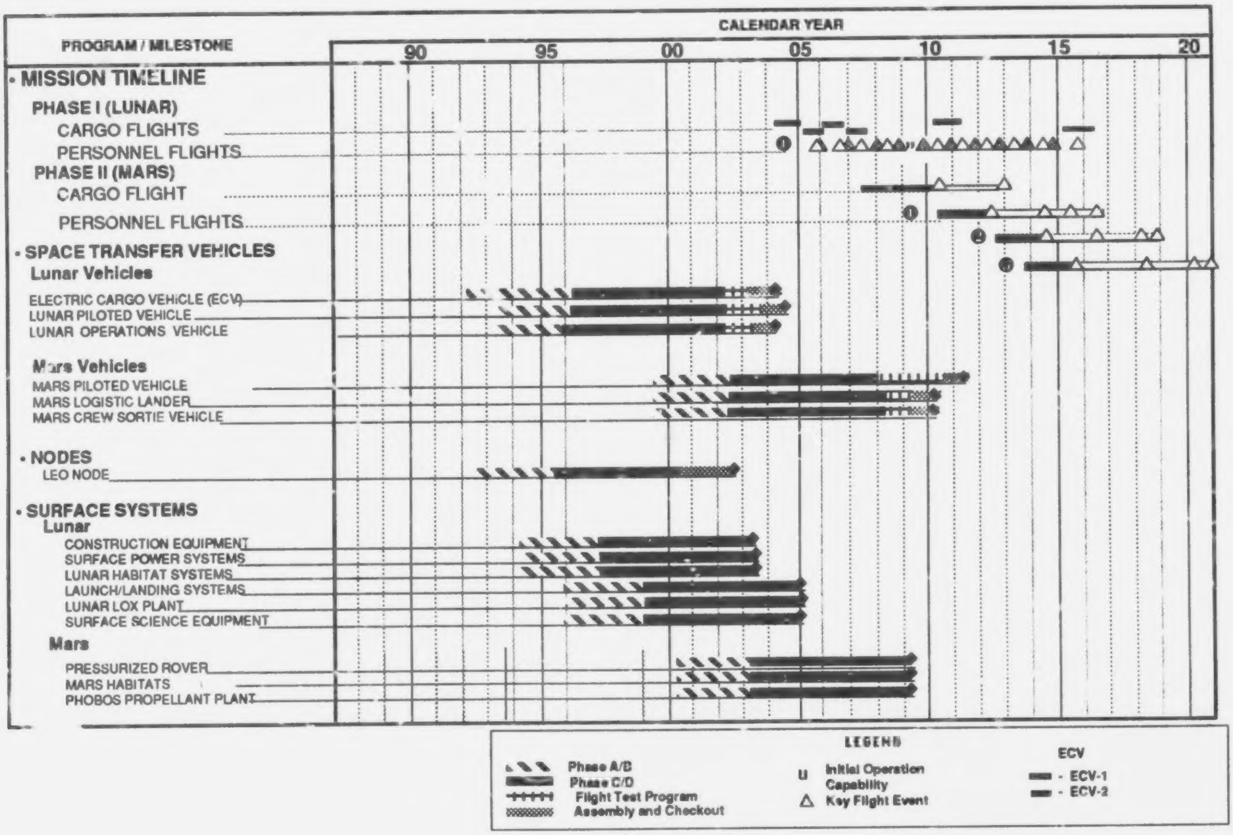


Figure 2.4.3-7.- Milestones for Lunar Outpost to Early Mars Evolution — mission requirements.

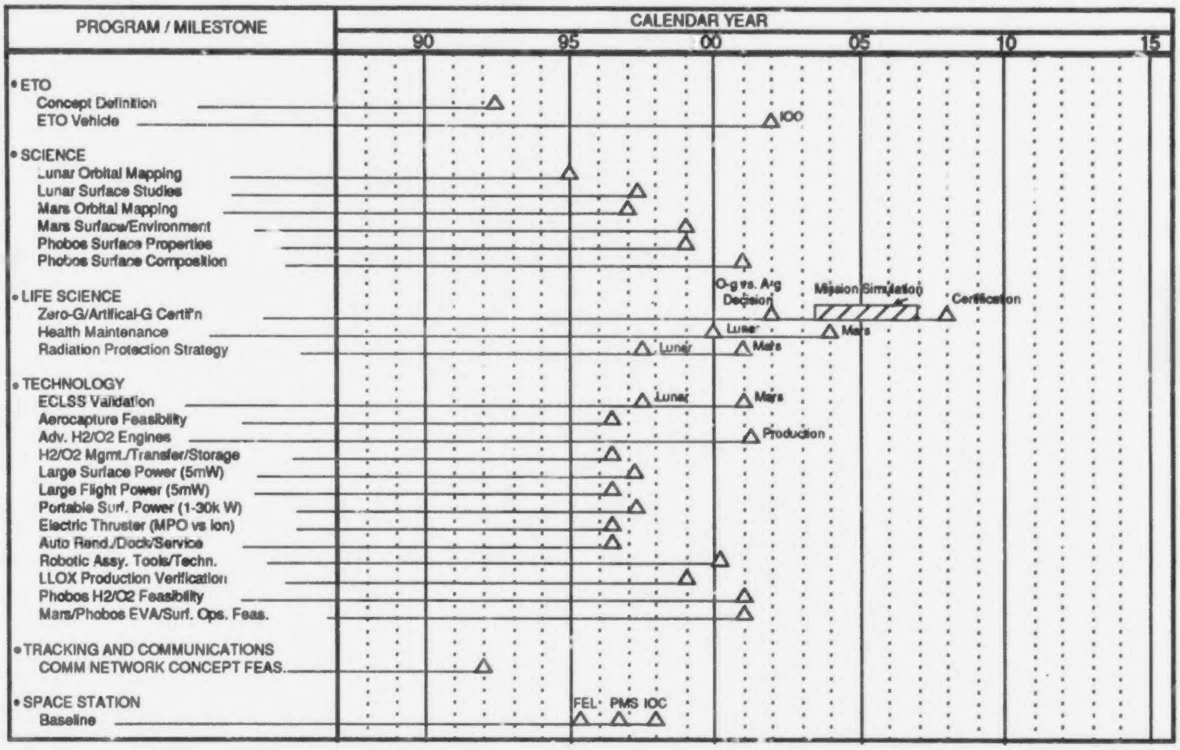


Figure 2.4.3-8.- Milestones for Lunar Outpost to Early Mars Evolution — prerequisite requirements.

Dry Mass	125.0 t
5 MWe Reactor, Engines	(75.0 t)
Tanks, Propellant Reserves (10% Propellant)	(19.0 t)
Payload Adaptor/Structure (5% Payload Capacity)	(31.0 t)
Propellant Type	Argon
Mixture Ratio	NA
Specific Impulse	6000 s
Payload Capacity	620
Crew Capacity	Unmanned
Propellant Capacity	190

Lunar Transfer Vehicle. The LTV transfers crew and some supplies from the LEO node to and from LLO. Once in LLO, the LTV makes rendezvous with the lunar personnel lander (LPL) or the ECV. The crew descends to the lunar surface in the LPL. The LTV is chemically propelled and uses an aerobrake for aerocapture into LEO from LLO.

Dry Mass (engines, structure, etc.)	7.9 t
Propellant Type	LOX/LH_2
Mixture Ratio	7/1
Specific Impulse	470 s
Payload Capacity (includes crew)	1.0 t
Crew Capacity	6
Propellant Capacity	18.5 t

Lunar Operations Vehicle. The LOV is the core component of the three lunar landing vehicles: the lunar cargo lander (LCL), the lunar personnel lander, and the lunar propellant tanker.

Lunar Cargo Lander. The LCL is the basic version of the LOV.

Dry Mass(engines, structure, etc.)	5.2 t
Propellant Type	LOX/LH2
Mixture Ratio	7/1
Specific Impulse	470 s
Payload Capacity (LLO to the Lunar surface)	40.0 t
Crew Capacity	Unmanned
Propellant Capacity	46.0 t

Lunar Personnel Lander. The LPL configuration is derived by attaching a crew module to the LOV and removing unnecessary propellant tankage.

Dry Mass(engines, structure, etc.)	7.5 t
Propellant Type	LOX/LH2
Mixture Ratio	7/1
Specific Impulse	470 s
Payload Capacity (includes crew)	1.0 t
Crew Capacity	6
Propellant Capacity	15.3 t

Lunar Propellant Tanker. The LPT configuration is derived by attaching lunar oxygen (LLOX) tanks to the LOV and adding propellant tanks.

Dry Mass (engines, structure, LLOX tanks etc.)	14.6 t
Propellant Type	LOX/LH2
Mixture Ratio	7/1
Specific Impulse	470 s
Payload Capacity (LLOX from the Lunar surface to LLO)	60.0 t
Crew Capacity	Unmanned
Propellant Capacity	52.0 t

Mars Piloted Vehicle. The MPV is used to transfer crew from LEO to the Mars system. The MPV consists of three primary elements: the interplanetary habitation module and life-support systems, the outbound stage, and the return stage. The vehicle is initially assembled in LEO on the ECV and then carried piggyback on the ECV to LLO. Here, lunar base cargo and hydrogen are exchanged for LLOX. The hydrogen is used in the LPT which delivers the LLOX to LLO for use in the outbound and return stages of the MPV. Once the MPV is filled with LLOX, the ECV spirals away from the Moon and puts the MPV into a high circular orbit about the Earth. The ECV returns to LEO, and the LTV, carrying the Mars crew, makes rendezvous with the MPV. The LTV returns to LEO and the MPV performs its Earth escape burn.

Dry Mass	188.0 t
(Interplanetary habitation, life support systems, structure, aerobrake, etc.)	(110.0 t)
Outbound Propulsion Stage	(63.0 t)
Return Propulsion Stage	(15.0 t)
Propellant Type	LOX/LH_2
Mixture Ratio	7/1
Specific Impulse	485 s
Crew Capacity	8
Propellant Capacity	522.0 t
Outbound Propulsion Stage	(420.0)
Return Propulsion Stage	(102.0 t)

Mars Logistics Lander. The MLL is used to deliver payloads from the ECV to the Mars surface. The MLL's are delivered by the ECV to the martian system where they separate from the cargo ship and use an aeroshield to shed velocity as they descend to the martian surface.

Dry Mass (engines, structure, aerobrake, etc.)	10.0 t
Propellant Type	LOX/LH2
Mixture Ratio	7/1
Specific Impulse	470 s
Payload Capacity (from Phobos to the Mars surface)	45.0 t
Crew Capacity	Unmanned
Propellant Capacity	20.0 t

Mars Crew Sortie Vehicle. The MCSV is used to transfer crews from the MPV to the Mars surface, Phobos, and Deimos. The MCSV's are delivered to the martian system by the ECV.

Dry Mass (engines, structure, aerobrake, etc.)	15.0 t
Propellant Type	LOX/LH2
Mixture Ratio	7/1
Specific Impulse	470 s
Payload Capacity (crew and manned module)	5.0 t
Crew Capacity	8
Propellant Capacity	60.0 t

2.4.5 Orbital Node Systems Definition

The results and descriptions presented in this section for Case Study 4 include no inputs from the node IA. Instead, only the initial conceptual MASE-derived inputs are discussed.

There are three primary node locations in Case Study 4: low Earth orbit (LEO), low lunar orbit (LLO), and low Mars orbit (LMO).

LEO Node. Accommodations in LEO to support Case Study 4 involve modifications to the Phase 1 Space Station Freedom and the construction of a vehicle assembly facility. In 2001, life sciences research capabilities at the Phase 1 Space Station Freedom are expanded with the addition of an outfitted human life sciences and animal/plant vivarium lab module as well as two solar dynamic generator modules, providing an additional 50 kW of power. These life sciences capabilities (45 t) support the development of advanced closed ecological life support systems (CELSS), medical techniques in space, and countermeasures for the effects of long-term exposure to the space environment. The focus of these efforts will be to develop the prerequisite knowledge for the long-duration missions to Mars and establish closure levels in the life support systems (lunar base and Mars base) that permit the initial colonization of the Moon and Mars.

The vehicle assembly facility (112 t) is launched to LEO and constructed in 2003 in preparation for the assembly, fueling, checkout, and refurbishment of the piloted LTV, the ECV, and the MPV. The primary components of this assembly facility include a habitation module for the assembly crew, a large vehicle assembly bay with a mobile servicing center and servicing arm, a propellant storage/fueling facility, a large hyperbaric airlock, and a service assembly command module. The vehicle assembly facility attaches to the phase 1 Space Station Freedom or, with some modifications, can act as an independent transportation node devoted to vehicle processing activities.

Once assembled, the ECV acts as a separate node in LEO for payload storage and checkout. ECV fuel storage, lunar vehicle fuel storage (primarily LH2), and ECV processing and maintenance. Servicing crews are stationed at the vehicle processing facilities and are ferried to and from the ECV only when necessary.

LLO Node. No permanent node exists in LLO for Case Study 4. Instead, all operations in LLO are carried out in conjunction with an ECV, which acts as a mobile node when at the Moon. The majority of the activities associated with the lunar operations vehicles (i.e., fueling, processing, checkout, and maintenance) are performed on the lunar surface. Activities in LLO focus on payload transfer and LH2 fueling. The ECV literally provides the framework (e.g., a servicing arm and power supply) to facilitate these operations.

LMO Node. Phobos and Deimos provide two extremely attractive natural nodes in orbit about Mars. Phobos is sufficiently small to allow a vehicle to dock with it rather than actually landing. Additionally, for local Mars operations, Phobos is lower in the martian gravity well than Deimos, allowing for smaller surface-to-orbit vehicles. Case Study 4 takes advantage of these features by using Phobos as a natural node in LMO. The first Mars ECV arrives in the martian system and docks with Phobos awaiting the arrival of the first MPV.

2.4.6 Planetary Surface Systems Description

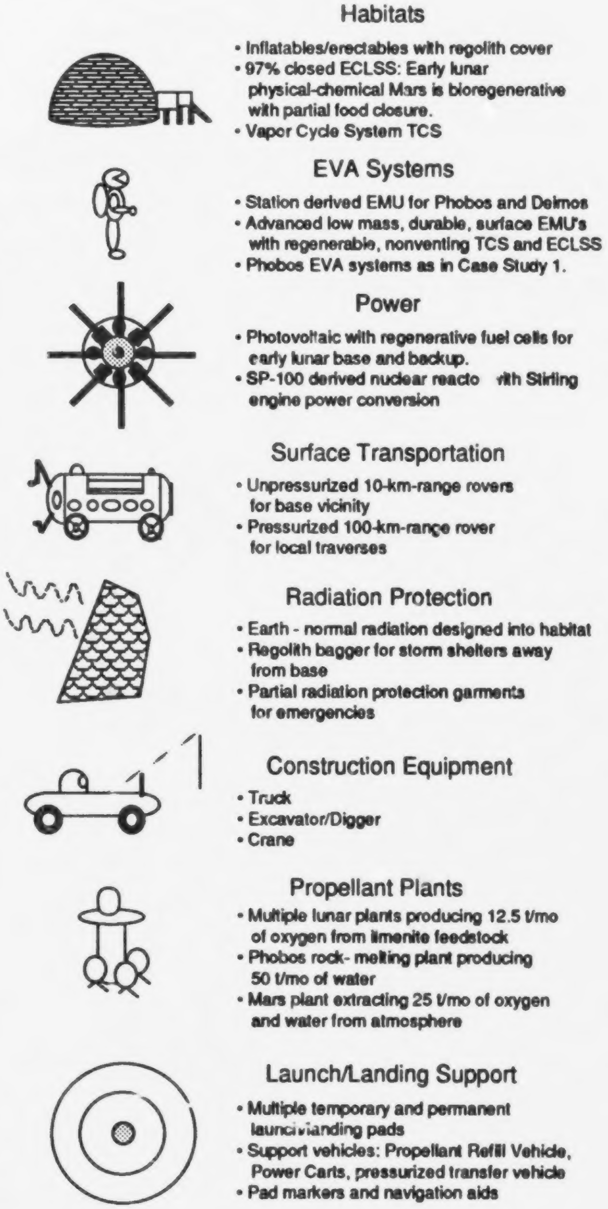
The lack of early-on maturity of Case Study 4 was an impact primarily in the areas of transportation and node requirements with a lesser impact on surface system requirements. Consequently, the following input from the Surface Systems IA formed the principal basis for the FY 1988 synthesis activity.

Case Study 4 establishes significant facilities on the lunar surface that will teach how to live and work on a planetary surface. This knowledge is used to establish a human presence on Mars. In addition, materials are produced from the Moon, Mars, and Phobos to support these endeavors.

2.4.6.1 Elements and Systems Description

Figure 2.4.6-1 summarizes the surface elements identified to support Case Study 4. In addition to those identified in the previous case studies (EMU's, Phobos EVA systems, construction equipment, regolith baggers, unpressurized rovers), the list includes many new elements. The surface habitats involve three major life-enabling components: structure, environmental control and life support system (ECLSS), and thermal control system (TCS). The ECLSS is substantially closed to reduce the logistics strain of continuous occupation. A pressurized rover permits extended traverses. Plants use local resources to produce substantial amounts of rocket propellants. Increased power needs are provided by a megawatt-class nuclear power plant.

Figure 2.4.6-2 depicts a concept for the lunar base layout. Primary power is provided by a nuclear plant whose reactor core is shielded by burying it in regolith allowing some freedom to place it near habitat and laboratory areas. Oxygen plants are located some distance away for safety and to isolate dust and contaminants. The liquid oxygen product is stored in buried tanks to facilitate



Habitats

- Inflatables/erectables with regolith cover
- 97% closed ECLSS: Early lunar physical-chemical Mars is bioregenerative with partial food closure.
- Vapor Cycle System TCS

EVA Systems

- Station derived EMU for Phobos and Deimos
- Advanced low mass, durable, surface EMU's with regenerable, nonventing TCS and ECLSS
- Phobos EVA systems as in Case Study 1.

Power

- Photovoltaic with regenerative fuel cells for early lunar base and backup.
- SP-100 derived nuclear reactor with Stirling engine power conversion

Surface Transportation

- Unpressurized 10-km-range rovers for base vicinity
- Pressurized 100-km-range rover for local traverses

Radiation Protection

- Earth - normal radiation designed into habitat
- Regolith bagger for storm shelters away from base
- Partial radiation protection garments for emergencies

Construction Equipment

- Truck
- Excavator/Digger
- Crane

Propellant Plants

- Multiple lunar plants producing 12.5 t/mo of oxygen from ilmenite feedstock
- Phobos rock- melting plant producing 50 t/mo of water
- Mars plant extracting 25 t/mo of oxygen and water from atmosphere

Launch/Landing Support

- Multiple temporary and permanent launch/landing pads
- Support vehicles: Propellant Refill Vehicle, Power Carts, pressurized transfer vehicle
- Pad markers and navigation aids

Figure 2.4.6-1.- Summary of planetary surface system elements for Case Study 4.

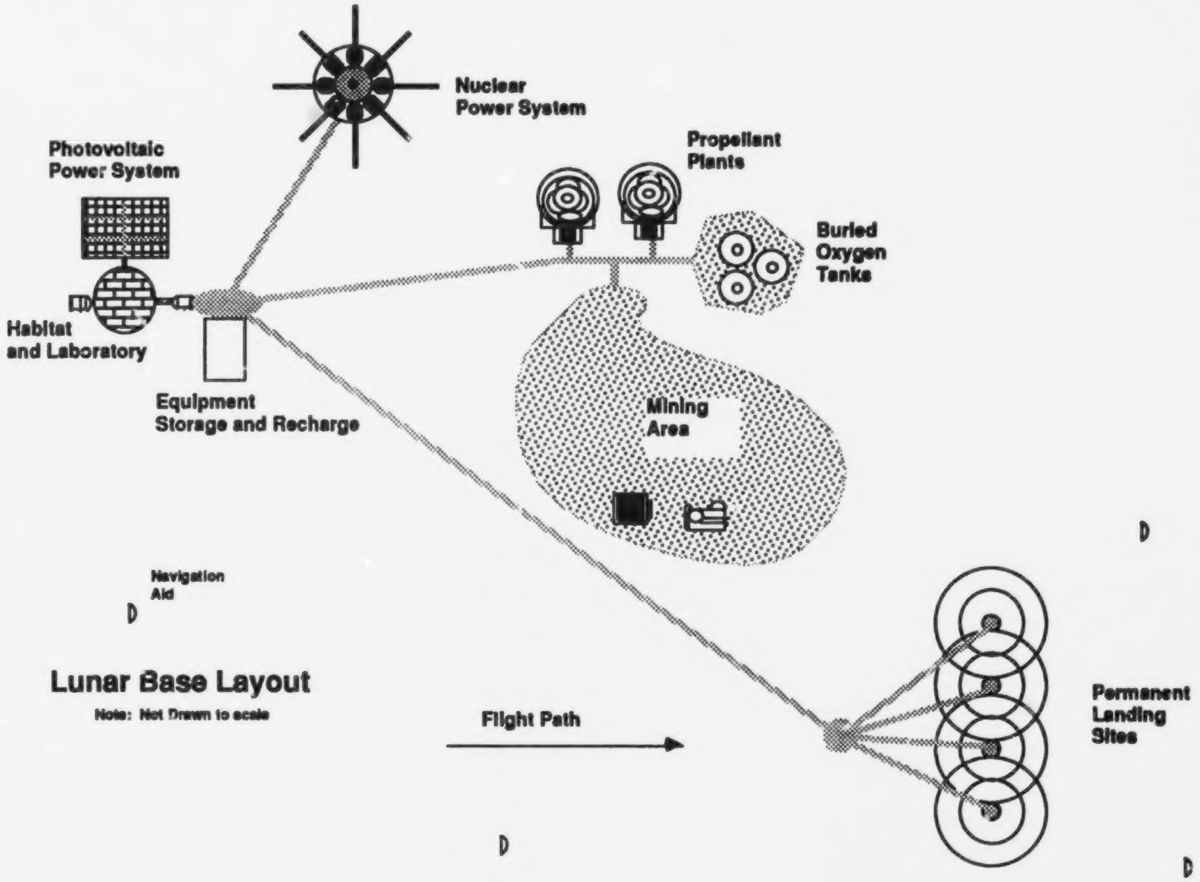


Figure 2.4.6-2.- Layout concept for lunar outpost.

cooling. A permanent landing/launch pad area lies some kilometers from the base to isolate debris lofted by rocket exhaust. Various navigation aids lie along the lander flight path. Support equipment provides services such as refueling and auxiliary power to landers while at the base. Improved roadways ease access between the major areas.

Inflatable/erectable habitat structures are chosen over modules since they provide more volume for a given mass. The inflatable, depicted in figure 2.4.6-3, consists of a spherical pneumatic envelope around a structural cage that supports floors, walls, and equipment. The cage also supports the envelope if pressure is lost. The design assumes that the habitat is inflated to standard sea-level pressure. A 2-m diameter vertical shaft provides access for crew and equipment. The habitat includes two airlocks, one of which is provided by a construction shack module that is connected to the inflatable by a flexible tunnel. The airlocks have front porches to facilitate cleaning and dusting off extravehicular mobility units (EMU's). The lower half of the habitat is buried below the surface and the top half is covered with bagged regolith for shielding from radiation and micrometeoroids. Burying substantially reduces hazards from external radiation. The envelope is a high-strength multi-ply fabric with an impermeable inner layer and a thermal coating outside. The structural frame is a cage of longitudinal and latitudinal curved beams that surround a combination of radial and concentric beams that support the flooring. A 16-m-diameter configuration has four floors and can house 12 crewmembers with total floorspace of 594 m^2. If made of a material similar to Kevlar-29, the envelope would be about 5 mm thick and would weigh about 3.3 t. The remaining mass totals about 16.3 t and includes the structural frame (9 t), floor (6 t) and walls (1.3 t).

Inflatables require more time to set up than pre-outfitted modules. In the current concept, a construction shack module lands near a hole that has been excavated with explosives. After shaping the hole, the inflatable is laid out, anchored, and erected. Covering with regolith is the most time-consuming task and is a prime candidate for automation.

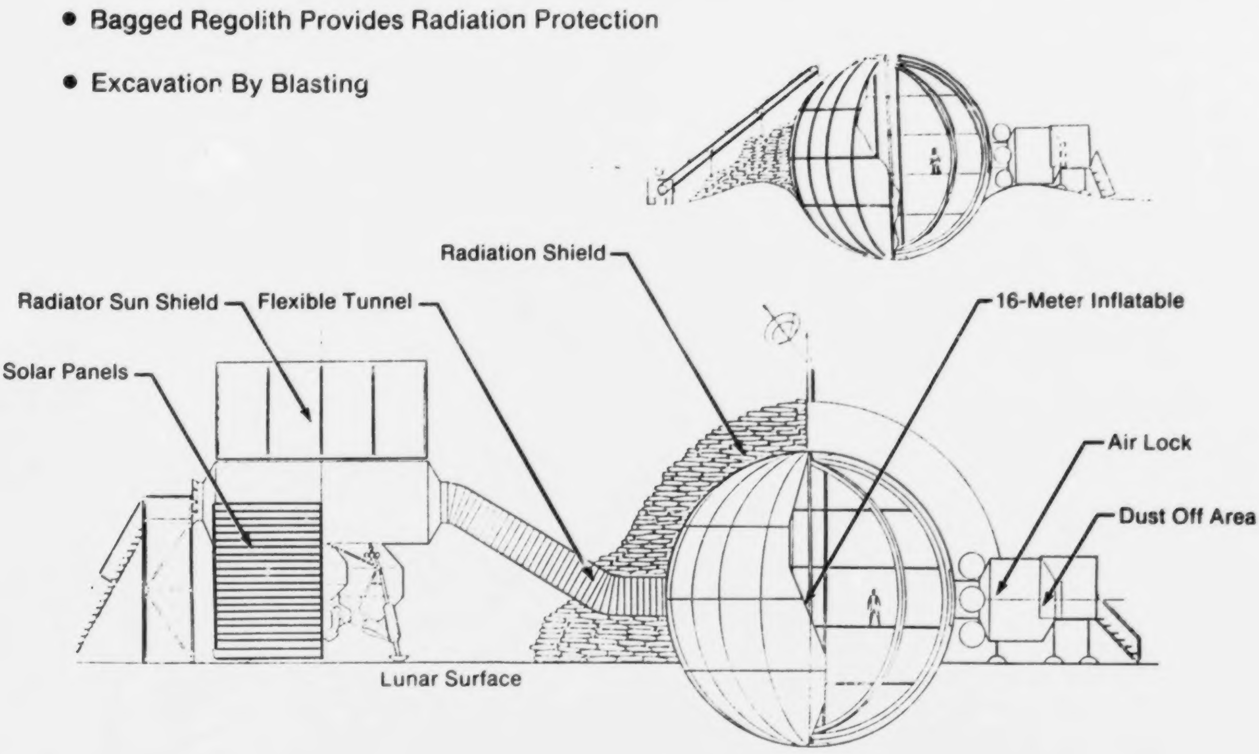


Figure 2.4.6-3.- Inflatable habitat with regolith cover.

A regenerative ECLSS is necessary for extended-duration missions to avoid prohibitive resupply logistics. The initial lunar ECLSS uses physical and chemical methods to regenerate oxygen and obtain 97 percent closure of the water cycle. ECLSS technology is assumed to evolve so that the Mars ECLSS is bioregenerative with partial closure of the food loop. The martian system uses local resources to make up water, oxygen, and nitrogen losses.

Total ECLSS closure is not feasible. Even the most optimistic estimates for a lunar base envision a bioregenerative system that recycles about 97 percent of the total mass with resupply of gasses lost through leaks and airlocks. The most important material loops are water, gasses, and food. Water is especially important because of its weight: resupply needs are about 0.93 t/yr per person with 90 percent closure and 0.28 t/yr per person with 97 percent closure. To achieve 97 percent closure involves recycling humidity condensates, wash and hygiene water, and urine. A major trade in designing an ECLSS is the cost of closure versus resupply. It is generally more economical to resupply trace substances than to recycle or reproduce them. With this in mind, the basic goals of regenerative ECLSS can be summarized as follows:

a. Keep material losses to a minimum.

b. Recover useful material from waste.

c. Reduce resupply logistics to a minimum.

The first two goals can be accomplished with physical and chemical means. Achieving the third goal requires post-Space Station Freedom ECLSS and/or biological systems.

The TCS provides for passive protection, acquisition, transport, and rejection of latent and sensible heat. Inside the habitat the major heat sources are metabolism and equipment. Since regolith provides good insulation from the surface environment, the major problem is heat rejection. To handle the drastic temperature variations in a lunar day, a cascaded vapor cycle system is envisioned. Two loops provide adequate heat rejection during the day when temperatures can reach 130°C and a bypass is provided to prevent over rejection at night when temperatures can fall below -150°C. The system provides final rejection temperatures of 43°C and 67°C to reject both the metabolic and equipment heat-loads during the day and provides a final rejection temperature of -11°C to reject the heat loads during the lunar night.

Radiation protection is a major concern for long-term habitation of extraterrestrial surfaces. The major hazards are from solar flares and lengthened exposure to galactic cosmic radiation (GCR). Solar flares occur sporadically and are roughly correlated with the sunspot cycle. GCR contains many more energetic particles than solar flares but at substantially lower fluxes. Solar flares can be lethal over short time periods whereas GCR presents a more long-term hazard. Shields of bagged regolith about 50-100 cm thick have been estimated to achieve a tolerable radiation environment for solar events. The shields also suffice for protection from micrometeoroids which generally penetrate only a few centimeters. Current GCR models are not yet adequate for predicting long-term shielding needs. With such coverings the habitats provide an adequate haven during a solar storm. EVA crew are at risk unless they can retreat to the habitat or some temporary haven. A regolith bagger provides for constructing temporary radiation shelters for crew when far from the base shelter such as during an extended traverse in the pressurized rover. Since the regolith bagging and stacking process can take a significant amount of time, it must be started somewhat before a solar storm.

Currently the ability to predict solar flares is somewhat limited, and warnings are best provided by surveillance of the sun. Warnings of solar storms may be as short as half an hour. Earth-based support can also be limited or nonexistent; for example, when Mars is on the opposite side of the Sun from the Earth. Improved ability to predict solar storms can reduce risks to crew since operations can be restricted during high alert periods. Radiation protection garments provide emergency partial protection when the crew does not have enough time to return to the habitat or construct a haven. The period of maximum flux of a solar storm is often on the order of a few hours. In such situations these garments give enough protection to limit exposure to tolerable levels for short periods of time. Such garments could consist of about 3 inches of multilayered carbon fiber and provide about 8 grams per square centimeter of shielding. This would reduce the dose rate of a solar flare by a factor of five to seven times that of an unshielded suit. During an event like the 6-hour peak of the August 1972 storm, one of the largest on record, they would allow for an emergency dose of about 10-15 rem as compared to 72 rem. However, they could not support an entire flare period but would give crew added time for more appropriate measures.

Including one propellant plant (150 t LOX/yr), base power needs are estimated to be in the 700-900 kWe range. Nuclear plants are favored at higher power levels because of their reduced mass. The lunar design envisions an SP-100-type reactor deployed in a cylindrical excavation with an aluminum bulkhead for protection from dust. This allows freer placement of the reactor relative to habitats and permits crew maintenance of radiator panels. Six high efficiency free piston Stirling engines running at 91.7 percent of capacity and two reserve engines ensure dependable power generation. Vertical spoke-wheel radiator panels and mercury heat pipes provide waste heat rejection. A PV/RFC power system provides for the early base and emergency backup. A nuclear power plant concept for Mars will be determined in FY 1989 studies.

The pad area is located several kilometers away to minimize blast effects. Analyses indicate that within 400 m, metal objects will experience significant pitting and glass surfaces will experience damage within 2 km. Permanent pads require surface stabilization such as gravel, paving tiles, or compaction. Gravel created as a byproduct of propellant production is a promising option. Pad markings and navigation aids help pilots and automated landers to find the pads and make precision landings. The devices envisioned are lightweight and contain a transponder, a visual marker, and a light. A retroreflector aids the use of a laser rangefinder. Since operation is infrequent and for short duration, power requirements are minimal. A number of specialized vehicles support pad operations. The construction crane is used to load and offload cargo to the truck. A propellant refill vehicle and power carts service the lander with fuel and auxiliary power.

The use of in situ resources offers great potential for bootstrapping and leveraging growth. FY 1988 activities focused on propellant plants for the surfaces of the Moon, Mars, and Phobos. Each is designed as a self-contained unit that includes its own power supply.

The lunar plant is baselined to use the hydrogen reduction of ilmenite process to produce oxygen from lunar regolith. Ilmenite is an iron titanium oxide whose two chief sources are high titanium basalts and mare soils. The ilmenite content of soils varies: about 7 percent by weight represents a typical value for rich deposits. Basalts can contain substantial ilmenite (the richest Apollo mare basalt samples contained about 33 percent by weight). Since the basalts require substantial crushing and grinding to release the ilmenite particles, the mare soil is preferred. Ilmenite reacts endothermically with hydrogen to produce water, iron, and titanium dioxide. Sufficient reaction rates require elevated temperatures. It has been reported that about 70 percent of the oxygen is removed after one hour at 1000°C. In the envisioned design, automated excavator vehicles mine the ore and deposit it into grizzly scalpers. A continuous conveyor carries the feedstock to the beneficiation process where the slightly magnetic ilmenite particles are removed with high intensity magnetic fields. If basalt feedstock is used, it is crushed, ground, and sorted before separation. Soil feedstock requires additional sorting and larger magnetic separators. Processing is done by feeding the ilmenite through low and high pressure hoppers into a

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three-stage fluidized-bed reactor. Most of the reaction takes place in the middle bed. Residual solids from the last bed are discarded through a solid gas separator after being used to preheat the material in the first bed. A solid state electrolytic cell dissociates the water into oxygen and hydrogen. The oxygen is liquefied for use as rocket propellant and the hydrogen is recycled. A pilot plant producing about 2 t/mo and powered by PV/RFC with a 35 percent duty cycle (daytime operations and hot standby at night) is estimated to weigh about 22.5 t. A 12.5 t/mo plant using nuclear power on a 90 percent duty cycle is estimated to weigh about 47.5 t.

The Phobos plant concept is sized to obtain 600 t/yr of water from rock and soil. The weak gravity of Phobos presents significant challenges but mining operations may prove more efficient than typical terrestrial ones. The shape and reflectivity of Phobos and Deimos suggest that they may be similar to carbonaceous chondritic asteroids. If so, they could consist of up to 20 percent water. The Phobos propellant plant design assumes a 5 percent water content and is based on a rock-penetrating prototype device that was developed at the Los Alamos National Laboratory. Laboratory and field tests with this prototype indicate that it is effective with most types and conditions of rock and soil. The plant, depicted in figure 2.4.6-4, uses a rock melter configured as a coring device. An impermeable glasslike lining forms in place around the borehole during penetration and seals in the released volatiles so that they do not escape into the surrounding

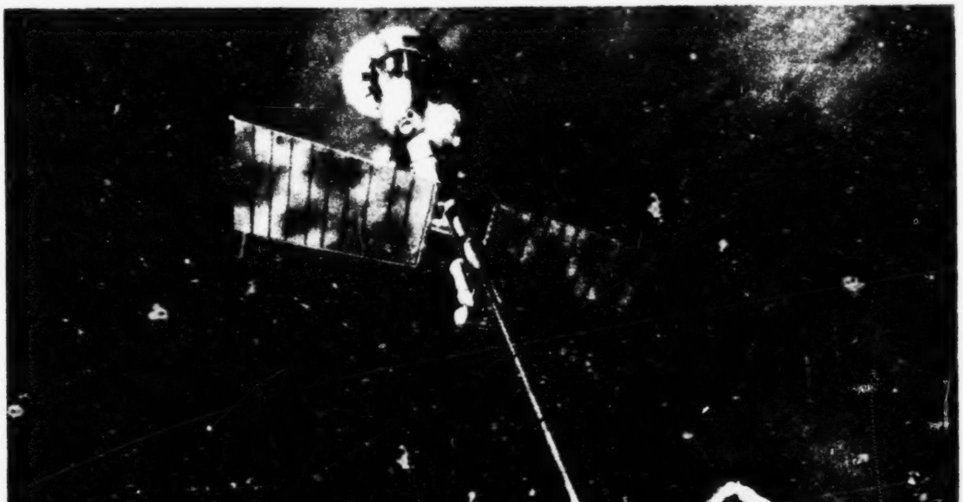

Figure 2.4.6-4.- Artist's rendition of a Phobos propellant plant.

porous rock. The released volatiles will probably contain such impurities as carbon monoxide, carbon dioxide, and hydrogen sulfide. Gross separation occurs when condensing water from the gases is emitted from the borehole. Absorption filters further purify the water which is then dissociated by electrolysis. The resulting oxygen and hydrogen are liquefied and stored. Between boring operations the plant makes short movements to new bore sites. This is accomplished by using legs with end-effectors after raising the plant with hydraulic jacks. The mass of a plant that extracts 600 t/yr of water is estimated at about 86 t with a power requirement of about 1067 kWe. The mass estimates include a self-contained nuclear power supply (20 t), radiation shielding, and habitat for crew (20 t).

The martian atmosphere consists primarily of carbon dioxide (about 0.955 molar fraction) and has a pressure of about 6 millibars at the surface. Water is present in the form of ice clouds and fogs. Large deposits of water ice occur at the polar caps. There is also a potential to find water ice in subsurface permafrost structures; however, it would probably require extensive drilling or mining to extract. The propellant plant design utilizes the carbon dioxide and water from the atmosphere. Oxygen is produced from the carbon dioxide by an electrolytic process. A blower forces martian air through a filter to remove particulates. The gas is compressed and preheated to 950 K and then enters an electrolytic unit that operates at 1273 K. Here the carbon dioxide dissociates into oxygen and carbon monoxide. Membranes in the electrolytic unit isolate the oxygen so that it can be removed in a relatively pure form. The unused exhaust gas is used to preheat the inlet gas before it is vented. The oxygen is liquefied and stored in buried tanks. The mass of a 300 t/yr plant is estimated at about 80 t with a 740 kWe power requirement. This estimate includes about 20 t for a nuclear power source.

Figure 2.4.6-5 depicts the growth of surface elements in this case study. In addition to the surface elements discussed above, the case study provides science payloads of about 100 t on the Moon and 50 t on Mars. A preliminary manifest includes the following:

a. Lunar base astronomy
b. Lunar life science lab
c. Lunar and martian geological traverses via rovers
d. Advanced materials processing
e. Martian teleoperated rovers
f. Martian geophysical/meteorological stations

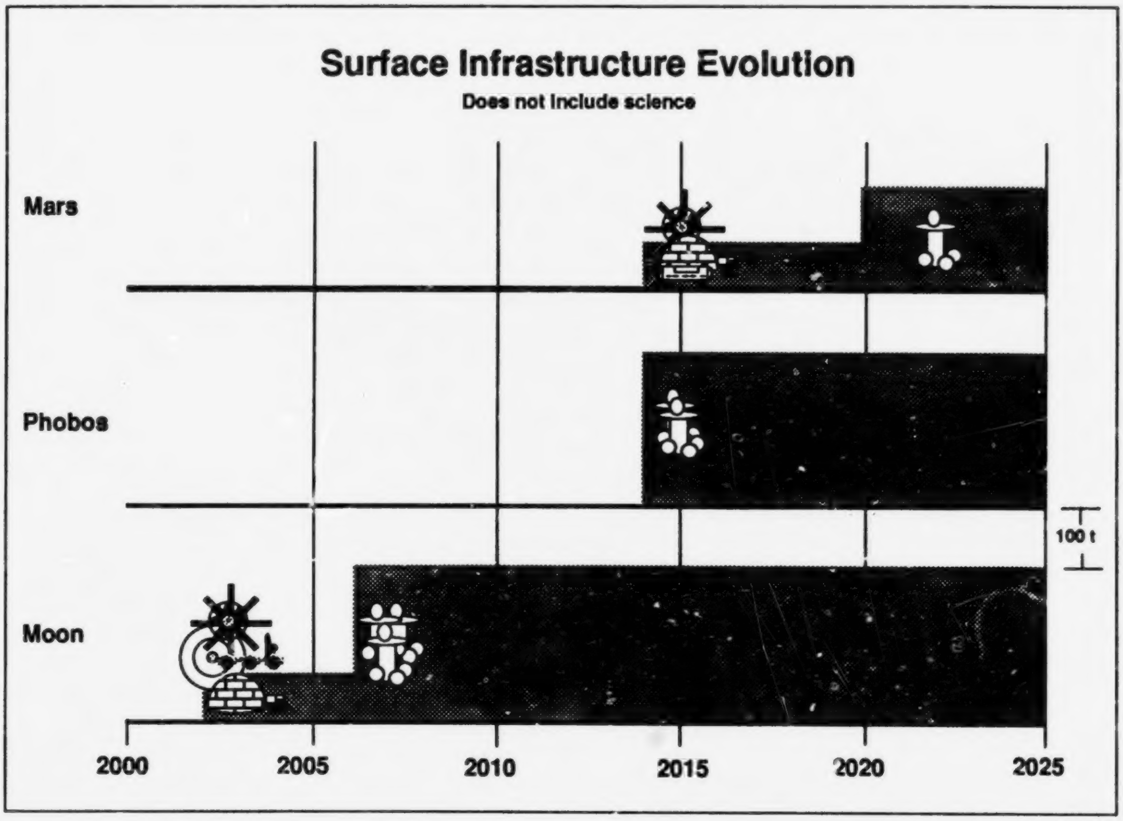

Figure 2.4.6-5.- Evolution of planetary surface infrastructure.

2.4.6.2 Technology Drivers

There are major technology needs in many areas: ECLSS; power generation and distribution systems; automation and robotics; construction techniques; in situ resource processing; long-life, low-maintenance mechanical systems; EVA systems and techniques.

Both physical/chemical and bioregenerative ECLSS technology is needed. Applications include fixed systems for habitats and portable systems for rovers and EMU's. Maximum closure (assuming use of local resources) is required. Portable systems for rovers and EMU's should be easily recharged at the base and provide for closure in the total system. The space station targets for about 95-97 percent water closure and recycling of oxygen. Space Station Freedom experience will be beneficial but should not be unduly counted on. Program constraints may limit the amount of closure obtained. Water closure has greater impact than oxygen recycling. There are also significant differences. Surface systems will operate in gravity environments different from those of an orbiting station. Architecturally, station modules are likely to form a more central core than an evolving surface system.

There are four basic problems with intermittent operation of ECLSS:

a. Complexity of startup/shutdown operations

b. Protection of ECLSS during down periods

c. Matching of process flow with use rates

d. Maintenance of sterility of process loops

Experience has shown that maintaining acceptable microbial conditions is very difficult — especially during down periods. Safety, reliability, and convenience are also important. Many candidate processes involve gasses such as hydrogen, oxygen, and ammonia. An advance concept operates at 250 atmospheres pressure and 670°F.

Adequate power is critical. Substantial, advanced bases will require nuclear plants in the 1 MWe class. The reactors should be safe, dependable, and easy to deploy and maintain. Other methods of power generation such as solar dynamic, advanced photovoltaics, and fuel cells have applications in rovers, isolated equipment, etc. In all methods, the construction and assembly time are important parameters that warrant further studies. Power distribution technology also needs attention. Preliminary estimates indicate that using cables to transmit power from a centralized power station can involve fairly high masses.

Current surface EVA and operations are inadequate to support this study. Current EMU's are too heavy and lack the mobility to support the frequent, long-duration tasks required. Advanced, durable EMU's that permit daily EVA's with limited preparation are needed. To meet planetary surface needs, Space Station Freedom EMU's require the following technology developments:

a. Durable, high-strength, low-mass materials

b. Improved low-mass thermal storage system

c. Improved long-life thermal-micrometeoroid garment materials; over-garment impenetrable to dust

d. Improved bearing seal systems

e. Improved low-torque, low-maintenance lower torso joints

f. Improved durability boot sole materials

g. Variable transmittance electrochromic systems for sun visors

Estimating the amount of crewtime needed to accomplish this scenario involves many fundamental uncertainties. Learning to "live and work" on extraterrestrial surfaces must be studied in more detail. Construction techniques and materials need further study and clearer definition. The actual capabilities and benefits of teleoperations and robotics for specific construction, maintenance, and operations tasks need much more definition.

Dependable surface transportation, especially long-range pressurized vehicles, require advances in several areas. The martian and lunar surfaces are harsh environments that require progress in the lubricants and materials used for mechanical interfaces. ECLSS should be lightweight, dependable, and easily serviced. Teleoperation either from Earth or the planetary vicinity can greatly enhance mission opportunities. Advanced locomotion techniques can aid in navigating rough terrain.

Processes and opportunities for in situ resources need critical study. Current baseline techniques for lunar propellant production require as much as 300:1 feedstock to product and have somewhat slow kinematics. Alternative techniques have been identified that promise higher yields and useful byproducts, but they generally involve more complicated chemistry or processes. Pilot systems for extraterrestrial applications can ensure operational technology when it is needed.

2.4.6.3 Systems Alternatives and Opportunities

A number of alternative structure and construction options have been suggested and deserve further study for base evolution. Candidates for large volume construction techniques include the following possibilities:

a. Prefabricated modules
b. Pneumatic/inflatable structures (air-inflated, air-supported)
c. Prefabricated frame structures
d. Tent structures
e. Lunar-assembled canopies
f. Craters (as shelters for habitats or as habitats)
g. Lava tubes (as shelters for habitats or as habitats)
h. Tunnels
i. Structures constructed of lunar-derived materials (sintered blocks, ceramics, glasses, metals)
j. Hybrid structures (combinations of the above)

The earlier items in the list are suitable for the initial base; the later items are probably best suited for a fairly advanced base with a developed infrastructure.

The TCS design requires substantial energy (about 40 kWe), but other approaches have severe disadvantages. Radiation rejection techniques with horizontal radiators and vertical sunshields appear applicable for power systems (with about 32°C rejection temperatures) that are beyond 15° of the equator. Movable canopies have been suggested for near the equator but they must be very high (estimates are over 58 m). Power needs can be reduced with reflective thermal blankets and elliptical mirrors that reduce the lunar infrared flux and the solar flux. However, at an active base there are concerns for the reflective surfaces because of dust and other contaminants.

The Mars diurnal temperature variation is between about 190 K to 240 K in the summer and near 150 K in the winter. Heat rejection problems are thus simpler than on the Moon and simpler TCS's are feasible.

Power systems present a variety of options. PV/RFC systems are suitable for lower power or daytime applications. They are safe, reliable, and relatively easy to install. Their major drawback is the mass required to supply power during the long (336-hr) lunar night. Nuclear power is preferred for the higher power levels of an advanced base. It offers substantial mass savings over PV/RFC systems for continuous operations at hundred kilowatt levels. The waste heat from nuclear plants provides an opportunity for additional energy. A simple deployable plant presents various placement and maintenance problems for an evolving advance base. Another basic trade is between centralized and distributed power strategies. Centralized systems appear most advisable for compact bases.

A number of enhancements, depicted in figure 2.4.6-6, for services at the pads are available. A tunnel ramp that

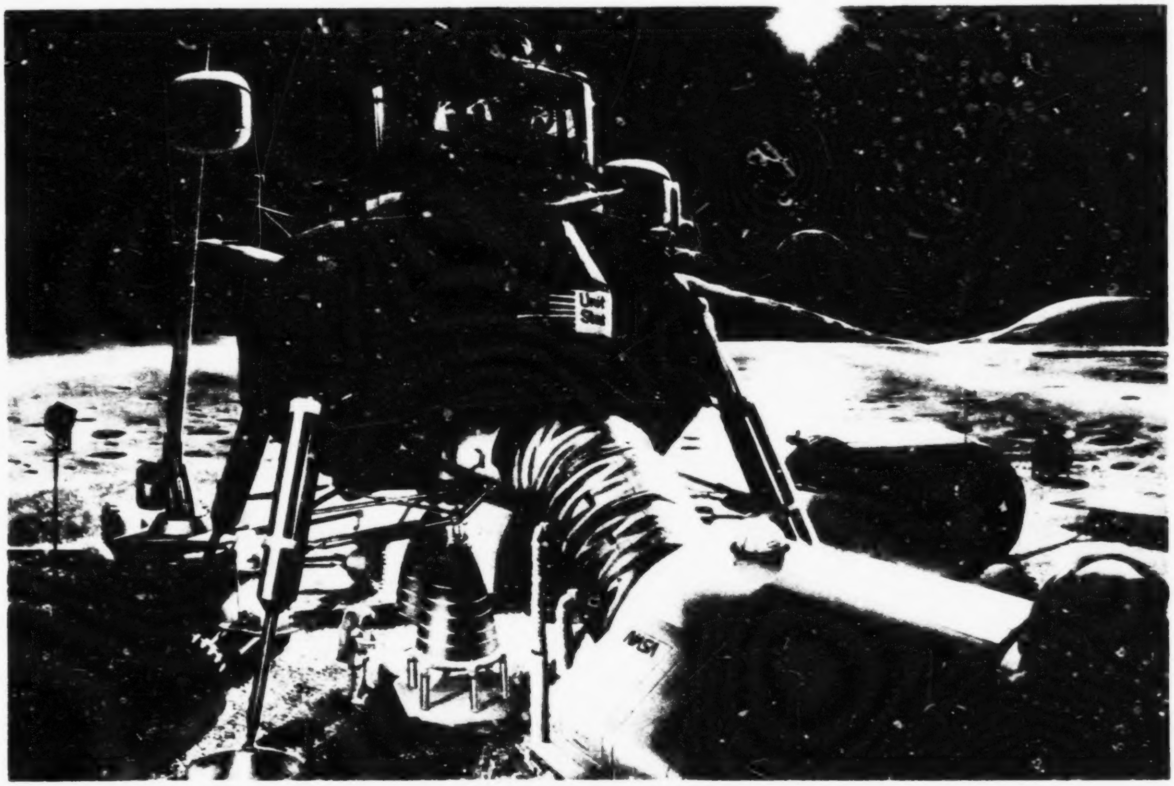

Figure 2.4.6-6.- Servicing a reusable lander on the lunar surface.

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provides access between the lander and a pressurized rover permits transfer of crew without EVA. Other support can include a supplemental cooling system, thermal and micrometeoroid blankets, and checkout and maintenance equipment. Portable and permanent blast barriers or blankets can alleviate blast problems.

Propellant production is an important initial step to using local planetary resources. There are many opportunities for growth and improvement. The hydrogen reduction of ilmenite technique extracts only a fraction of the oxygen available in lunar materials. Although the chemistry is simple, the kinematics is slow and the yield is low (an estimated one t of oxygen per 306 t of regolith feedstock). Numerous other options for lunar oxygen production have been studied to varying degrees. These include, to name a few, carbothermal reduction of oxides, fluorine exchange, hydrofluoric acid leach, and electrolytic reduction of oxide/caustic solutions. These vary in the complexity of the chemistry involved and the amount of oxygen and byproducts obtained. In addition to oxygen these processes can yield various metals byproducts such as iron, titanium, and aluminium. There are also opportunities for using excess and waste material for construction. For example, the baseline beneficiation process leaves a gangue consisting of mixed particles of pyroxene, plagioclase, and olivine minerals that could be sintered to form ceramic blocks for construction. Apollo regolith contains from 26 to 54 p/m of hydrogen in the form of molecular hydrogen and water. Moderate heating (600-900°C) releases the hydrogen that can be used for fuel or makeup.

2.4.7 Case Study Synthesis

2.4.7.1 Evaluation of Inputs

The accomplishment of objectives within a constant annual investment strategy is of fundamental importance to this case study. Specifically, the FY 1988 Exploration Requirements Document (ERD) ground rule coupled with the implementing Study Requirements Document (SRD) requirement placed a limit of 318 t per year as the allowable steady-state mass to low Earth orbit. The case study employed advanced technology to leverage mass reduction along with other approaches such as trajectory strategies to finally fit within the limits of the imposed constraint. This maturation process for the case study did not permit a full cycle for IA analysis for FY 1988. Consequently, the decision was made to conduct the synthesis based upon MASE-derived inputs for the final version of Case Study 4. The synthesis results will initiate the full cycle analysis for FY 1989.

Whereas the baseline for Case Study 4 employs nuclear electric propulsion, an independent indepth assessment (see section 5.2.3) by the Propulsion SAA (LeRC) explored the use of gas-core NTR technology. This technology, which could be available in the next several decades, offers promise of simultaneously reducing trip times, logistical complexity, and mass. Additional analyses will be conducted in FY 1989.

2.4.7.2 Principal Issues and Program Risks

Advanced technology is the two-edged sword for Case Study 4: although it holds the key to the accomplishment of the desired objectives, it holds considerable program risk. Perhaps the biggest programmatic risk is the early-on availability (within the first part of the decade after the turn of the century) of the electric cargo vehicle with its 5 MW_e nuclear power system and ion thrusters. Future versions of this case study may require introduction of this vehicle at a later date. Propellant production, storage, and transfer on the lunar surface and at Phobos require advanced technological development and are considered to be areas of high program risk. Other technology drivers and high risk areas are aerocapture systems, human performance degradation countermeasures, surface power systems (including nuclear), Phobos and Mars EVA systems and techniques, ECLSS closure, and advanced chemical propulsion. In addition to the risks associated with advanced technology, there is also the risk associated with the complexity and long duration of any manned Mars mission.

SECTION 3

Prerequisite Implementation Plans

As an integral part of its development and analysis of a range of alternatives for extending human presence beyond low Earth orbit (LEO), the Office of Exploration (OEXP) must integrate other NASA programs' functional capabilities and responsibilities that are prerequisites to the implementation of one or more exploration alternatives. Therefore, OEXP provided requirements for human exploration to the various program offices in the form of a Prerequisite Requirements Document (PRD). The affected NASA program offices (codes M,E,R,T, and S) responded to the PRD by providing the OEXP with their respective prerequisite requirements implementation plans. Each NASA program code's plan summarizes the strategy for accommodating OEXP requirements for which that office is responsible. Plans include the identification of projected transportation and support hardware capabilities, science and applications data required as a precursor to human planetary missions, technology research and development programs, flight test programs, communications support strategies, and methods of element assembly support in LEO. The prerequisite program implementation plans from each code are presented in sections 3.1 through 3.6.

Each program also provided data within its plan on the hardware and technology definition and development milestones necessary to provide the required capabilities. The OEXP, through the Mission Analysis and System Engineering (MASE) function, took the data from each code's report and created an integrated program definition and development schedule for each case study. These integrated schedules appear in Section 3.7.

3.1 EXPLORATION IMPACTS TO EARTH-TO-ORBIT (ETO) TRANSPORTATION

3.1.1 Role of the Office of Space Flight (OSF) in Manned Exploration

3.1.1.1 Roles, Responsibilities, and Strategy

OSF (code M) responsibilities are to develop an integrated transportation strategy to reflect total national needs; to conduct concept definition, advanced development, and systems definition studies for safe, reliable, and cost-effective transportation elements; and to develop the transportation elements on a schedule that facilitates mission requirements. The human exploration missions will have a major impact on the magnitude of the civil transportation requirements. In the human exploration era, the size and quantity of vehicles required for these missions will be major drivers for transportation system development. Near-term transportation strategy will be affected by the need to incorporate evolutionary characteristics into vehicle designs and to gain operational experience relevant to the extended missions.

As the OEXP defines the alternative mission scenarios for extending human presence beyond LEO and identifies specific requirements, OSF will assess the impact and implications on the transportation system and provide feedback to OEXP. In the near term, OSF will develop concepts and designs specifically for the earth-to-orbit (ETO) transportation elements, although evolutionary provisions to facilitate growth into the regime beyond LEO for human exploration initiatives will be included in the design of all ETO transportation elements. OSF will also conduct concept definition studies for an interim space transfer vehicle reflecting unmanned planetary, geosynchronous, and potential Department of Defense requirements. OEXP will take the lead for study of in-space transportation options and concepts for human exploration. As OEXP narrows the options for in-space transportation architecture and concepts and derives the mission requirements, the responsibility for indepth definition, design, and development of transportation elements will undergo transition to OSF.

Assessing the degree of fit between current OSF planning and the OEXP requirements is difficult at this time. Conceptually there is good fit; however, concerns arise as to the schedule requirements, number of launches, and size of vehicles. The ground facilities to handle propellants and launch on the required schedules are a potential problem. Transportation planning has assumed increased emphasis on orbital operations. Scenarios that place almost all the burden on ground facilities will require further assessment. The impacts of all the operational requirements will be analyzed in more depth during the next study cycle.

3.1.1.2 Programmatic Impacts

In the near term, the usual OSF transportation planning functions will be impacted primarily by the magnitude of the requirements and number of options to be assessed. The processes of evaluating requirements, developing concepts, and conducting trade studies are standard procedures; however, the number of alternatives to be assessed will be much higher than usual. Also, a faster "turnaround time" will be required to provide feedback to exploration planners. In most cases the transportation requirements from the exploration scenarios are consistent with the transportation planning that is already in place. An exception is the very large ("magnum") vehicle required by one of the OEXP guidelines for Case Study 1; an indepth study of this vehicle is beyond the scope of current OSF planning.

3.1.2 Case Study Needs Assessment[1]

This section summarizes the OSF response to the case study requirements provided by OEXP for ETO transportation. OEXP provided specific ground rules for some of the ETO transportation options; therefore, the size or type of some transportation elements shown here is not necessarily the optimum solution that could accommodate all future alternatives, but the solution responsive to the ground rules.

3.1.2.1 Mass-in-LEO Sensitivity to Launch Date

Planning for the delivery of the ETO cargo for planetary missions is complicated by the fact that LEO mass requirements vary as a function of planetary mission launch date. This is due primarily to the additional propellants required to deliver the payloads from LEO to the planetary destination when the Earth and planetary orbital geometries are not favorable. The ratio of the mass required in LEO to the mass finally delivered to the planetary vicinity can vary by a wide margin as depicted in the upper curve of figure 3.1.2-1 for Case Studies 1 and 2, the Phobos and Mars expeditions. If trip times are not a concern, more efficient trajectories are available, as shown in the lower curves. However, the piloted missions need short trip times to minimize the effects of

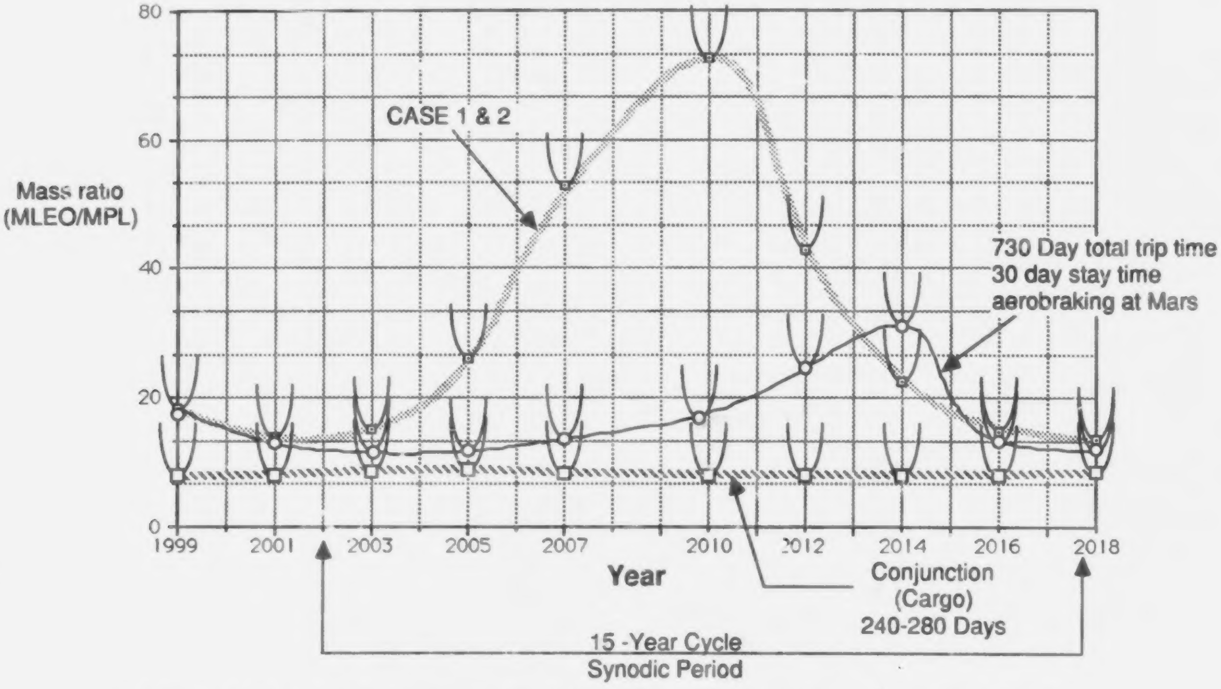

Figure 3.1.2-1.- Mars mission mass-in-LEO sensitivity to launch opportunity and trip time.

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[1] Section 3.1.2 was prepared using original case study initialization parameters provided by OEXP. During subsequent iterations in the synthesis process, several key parameters were modified to accommodate long-term NASA program plans and schedules, most notably the slip by one launch opportunity (26 months) of the first and subsequent missions in the Human Expeditions to Mars Case Study. Due to changes in relative planetary positions, this slippage significantly affected mass-to-LEO requirements for the third mission as it was originally designed; to offset this effect, trajectory and mission duration parameters were modified for the third mission only. OXEP, using original inputs provided in the OSF report, has re-sorted data within this section as required to approximate ETO requirements which will satisfy the new Mars mission launch dates, also included in this section. There remain minor discrepancies between the launch manifests presented in this section and those manifests used in section 2; however, conclusions related to ETO needs were unaffected. OSF and OEXP will reanalyze these data during the next reporting period and modify ETO requirements as necessary.

interplanetary travel on the crew. The variation in total ETO mass dictates that either the scheduled launch date must be met or the vehicles must be oversized to accommodate a possible slip in the launch date. The vehicle sizing shown here is for the specific launch dates designated in the case studies.

3.1.2.2 Orbital Assembly Operations

The exploration missions will require onorbit assembly. The question is, "how much?" A trade inherent in preparing for the exploration missions is orbital assembly versus launch vehicle size. Definitive data are not yet available. However, trends indicate that as launch mass and volume increase, opportunities are enabled to reduce the complexity of orbital operations. This is shown schematically in figure 3.1.2-2. In the ultimate case, all of the payload mission equipment could be risked on one ETO launch for which there is little experience base. Concurrently, the complexity of ground operations would also increase. At this time the experience base for ground operations is much more extensive than for orbital operations. With the advent of a space station, orbital operations will be much more common. This hands-on experience will be a valuable precursor for human exploration missions, although the size of the elements required for the exploration missions will be much larger than for any previous endeavors.

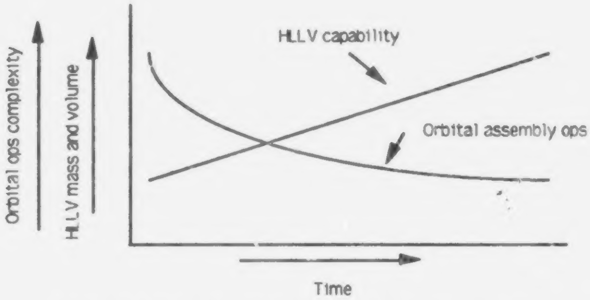


Figure 3.1.2-2.- Orbital assembly operations for large payloads.

3.1.2.3 Assessment of the Case Studies

The ETO requirements for all four cases were analyzed by itemizing the elements of the space transportation vehicle plus associated payloads and propellants, segregating these by increments of volume and mass, and manifesting them on the ETO launch vehicles. Additionally, two special cases were analyzed for Case Study 1, the Phobos Expedition, to conform to OEXP ground rules that the number of ETO launches should be minimized. For the initial analysis of all case studies, all personnel flights were assumed to be on the Space Transportation System (STS). The numbers of heavy lift launch vehicles (HLLV's) required for the cargo launches are shown in table 3.1.2-I as a function of launch vehicle mass capability. The analysis revealed that propellant delivery flights typically are mass-limited, whereas other flight are usually volume- or length-limited. The date that an HLLV must be operational in order to support a

TABLE 3.1.2-I.- SUMMARY OF ETO LAUNCH REQUIREMENTS

HLLV Capability	Case 1A	Case 1B (see note 1)	Case 1C (see note 2)	Case 2	Case 3	Case 4
100 - 200 t		10				
90 - 100 t	19			65	2	
80 - 90 t	2				4	4-6/yr
70 - 80 t	2			10	10	
60 - 70 t	1			5		
50 - 60 t				4		
40 - 50 t	1			4	6	3
30 - 40 t						
20 - 30 t						
10 - 20 t				3		
HLLV TOTAL	25	10	2	95	18	TBD
STS	5	3	1	19	10	TBD

Note 1. Case 1c is accomplished with only 2 ETO launches using a 1360-t "magnum HLLV." Launch of the cargo vehicle delivers 467.0 t. Launch of the piloted vehicle delivers 1311.3 t.

Note 2. Case 1b utilizes a 200-t "very large HLLV"

Case 1a: Payloads ranged from 42.2 t to 91 t
Case 1b: Payloads ranged from 103.4 t to 200 t
Case 1c: Payloads ranged from 467.0 t to 1311.3 t
Case 2 : Payloads ranged from 10.5 t to 91 t
Case 3 : Payloads ranged from 49.3 t to 90.9 t
Case 4 : 91 t requirement assumed

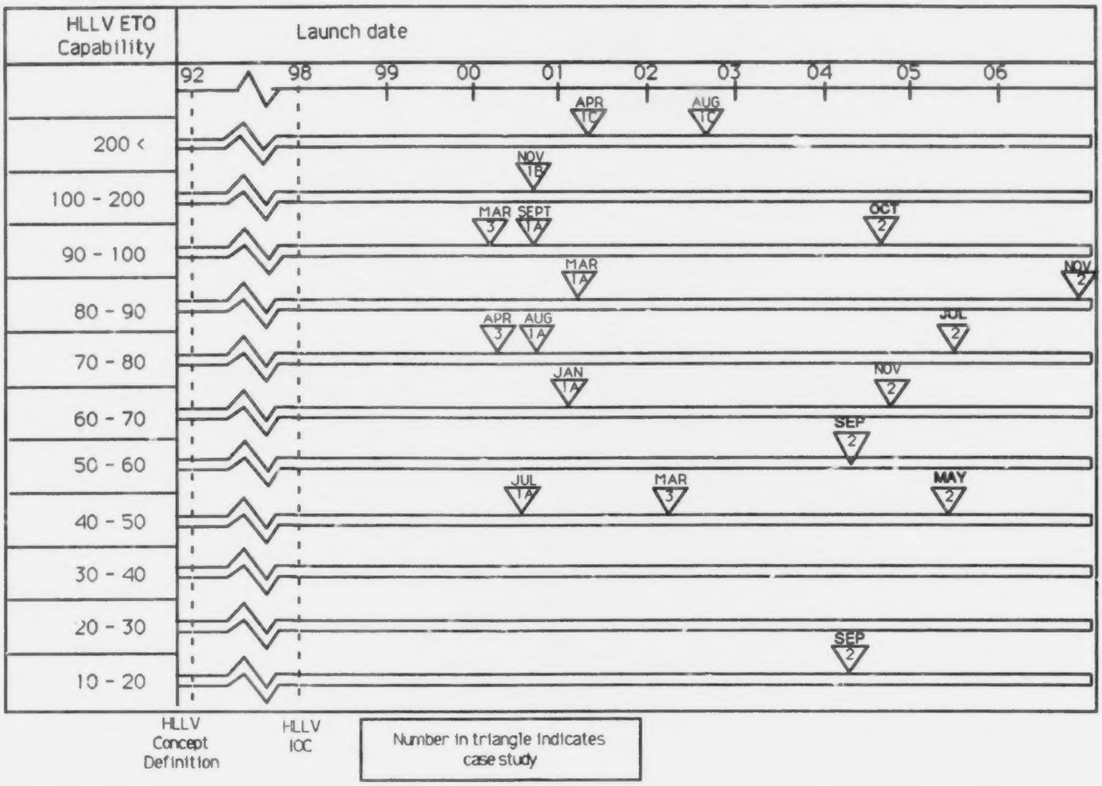


Figure 3.1.2-3.- Summary of case study HLLV need dates.

specific option is shown in figure 3.1.2-3. If the HLLV is required to support precursor missions, the need dates may be earlier; this analysis will be completed in the next study cycle. Case Study 2 is the Human Expeditions to Mars; Case Study 3 is Lunar Observatory. Case Study 4 is Lunar Outpost to Early Mars Evolution.

The STS flights in the tables do not take into account the possibility of sharing flights with space station crew rotation. This would significantly reduce the estimates contained in this report. Forecasting space station traffic models is beyond the scope of this activity.

Human Expedition to Phobos (Case Study 1). The major objective of the Phobos expedition is to achieve an early human presence in the neighborhood of Mars. Compared to the Mars expedition, this mission has reduced hardware complexity and reduced total mass. The basic characteristics of the mission are listed in table 3.1.2.-II. Additionally, major OEXP guidelines for the mission analysis are (1) all-propulsive braking of the space transfer vehicles, (2) minimum onorbit assembly and checkout, and (3) a sprint trajectory for short interplanetary mission duration. In this report, guideline number 2 (minimum onorbit assembly) is reflected in Case Studies 1B and 1C. Table 3.1.2.-III lists the manifest for Case Study 1A that uses a 91-t vehicle, consistent with the

TABLE 3.1.2.-II.- HUMAN EXPEDITION TO PHOBOS — MISSION DESCRIPTION

Case Study Descriptor	Human expedition to Phobos
• Transportation	
- Trajectory profile	Cargo: minimum-energy Crew: sprint
- Number of flights	1 Cargo, 1 crew
• Crew size	4
• Crew total sortie time	440 days
• Surface staytime	20 days at Phobos
• EVA's (per mission)	4 EVA's at Phobos
• Propellant production	N/A
• Cargo mission	
- LEO departure date	Feb 2001
- Phobos arrival date	Oct 2001
• Piloted mission	
- LEO departure date	August 2002
- Phobos arrival date	May 2003
- Earth return date	October 2003

TABLE 3.1.2.-III.- ETO MANIFEST — HLLV LAUNCHES
FOR CASE STUDY 1A (91-t VEHICLE)

	Launch	Date	Item	Mass (t)
			Mars Cargo Vehicle (MCV) Components Launches for Assembly	
1.	HLLV-1	Jul '00	Mars transfer vehicle (MTV) Phobos excursion vehicle	44.9
2.	HLLV-2	Aug '00	MOCS & MOOS	72.1
3.	HLLV-3	Sep '00	TMIS tank #1	91.0
4.	HLLV-4	Oct '00	TMIS tank #2	91.0
5.	HLLV-5	Nov '00	TMIS tank #3	42.2
6.	HLLV-6	Dec '00	TMIS tank #4 (small tank) plus TMIS engine, PA for top-off prop	74.5
7.	HLLV-7	Jan '01	TEIS (fully loaded)	64.2
			Mars Spaceship (MSS) Components for Assembly in LEO	
8.	HLLV-8	Mar '01	MTV, minus MOCS and MOO	81.1
9.	HLLV-9	Apr '01	MOCS + MOO tank #1 plus engine	91.0
10.	HLLV-10	May '01	MOCS + MOO tank #2 plus engine	91.0
11.	HLLV-11	Jun '01	MOCS + MOO tank #3	91.0
12.	HLLV-12	Jun '01	MOCS + MOO tank #4 plus engine	91.0
13.	HLLV-13	Jul '01	TMIS tank #1	91.0 each
	thru	thru	thru	
24.	HLLV-24	Apr '02	TMIS tank #11	
25.	HLLV-25	Apr '02	TMIS tank #12 plus engine (incl. 10 t for TMIS, MOCS top-off)	82.9

TABLE 3.1.2 -IIIa.- ETO MANIFEST — STS LAUNCHES
FOR CASE STUDY 1A

	Launch	Date	Item
1.	STS-1	Dec '00	Assembly and integration of cargo vehicle
2.	STS-2	May '01	Assembly and integration of piloted vehicle
3.	STS-3	Sep '01	Crew rotation
4.	STS-4	Jan '02	Crew rotation
5.	STS-5	Jun '02	Flight crew delivery and return of checkout crew

projected size of the advanced launch vehicle. Table 3.1.2.-IV lists the manifest for a 200-t vehicle. Table 3.1.2.-V shows the vehicle capabilities required if the earth-to-orbit launches are limited to one each for the Phobos cargo vehicle and piloted vehicle, assuming that the "all-propulsive space transfer vehicle" requirement is operative. (Note: After preparation of this report, OEXP added an aerobraking option to the "all-propulsive" guideline. The total mass requirements for this optional mission will be lower and will be analyzed in the next study cycle.)

Human Expeditions to Mars (Case Study 2). The major objective of the Mars expedition is to establish a permanent human presence on the surface of Mars. The basic characteristics of the missions are listed in table 3.1.2-VI.

TABLE 3.1.2.-IV.- ETO MANIFEST — HLLV LAUNCHES
FOR CASE STUDY 1B (200-t VEHICLE)

Launch	Date	Item	Mass (t)
Mars Cargo Vehicle (MCV) Components Launches for Assembly			
1. HLLV-1	Nov '00	MTV	163.6
2. HLLV-2	Dec '00	TMIS, partial	200.0
3. HLLV-3	Jan '01	TMIS, complete	103.4
Mars Spaceship (MSS) Components for Assembly in LEO			
4. HLLV-4	Dec '00	MTV	124.4
5. HLLV-5	Jan '01	MOCS partial	200.0
6. HLLV-6	Mar '01	MOCS complete + partial TMIS	200.0
7. HLLV-7	Feb '01	TMIS partial	200.0
8. HLLV-8	May '01	TMIS partial	200.0
9. HLLV-9	Apr '01	TMIS partial	200.0
10. HLLV-10	Jun '01	TMIS complete	200.0

TABLE 3.1.2.-IVa.- ETO MANIFEST — STS LAUNCHES
FOR CASE STUDY 1B

Launch	Date	Item
1. STS-1	Jan '01	Assembly and checkout of cargo vehicle
2. STS-2	Mar '01	Assembly and checkout of piloted vehicle
3. STS-3	Jun '01	Flightcrew delivery and return of checkout crew

TABLE 3.1.2.-V.- ETO MANIFEST —
HLLV LAUNCHES
FOR CASE STUDY 1C (TWO LAUNCHES)

Launch	Date	Item	Mass (t)
Mars Cargo Vehicle (MCV) Components Launches or Assembly			
1. HLLV-1	Jan '01	MCV	467.0
Mars Spaceship (MSS) Launch			
2. HLLV-2	Jun '01	MSS	1311.3

TABLE 3.1.2.-Va.- ETO MANIFEST —
STS LAUNCHES FOR CASE STUDY 1C

Launch	Date	Item
1. STS-1	Jan '01	Flightcrew delivery

TABLE 3.1.2.-VI.- HUMAN EXPEDITIONS TO MARS — MISSION DESCRIPTIONS

Case Study Descriptor	Human expeditions to Mars		
• Transportation	Cargo: minimum-energy		
- Trajectory profile	Crew: opposition-class		
- Number of flights	3 cargo, 3 crew		
• Crew size	8 (4 to Mars Surface)		
• Crew total sortie time	< 500 days		
• Surface staytime	20 days on Mars surface		
• EVA's (per mission)	4 EVA's at Phobos/Deimos 10 EVA's at Mars (Five 10-km unpressurized rover traverses)		
• Propellant production	N/A		
• Mission dates	Mission number		
- Cargo	1	2	3
- LEO departure date	9/'05	9/'07	10/'09
- Phobos arrival date	10/'06	9/'08	9/'10
• Piloted			
- LEO departure	12/'06	2/'09	1/'11
- Mars arrival	8/'07	10/'09	9/'11
- Earth return	3/'08	4/'10	5/'12

A staging node in LEO is required to support onorbit assembly and checkout of spacecraft and transportation elements.

Both artificial gravity and zero gravity options are addressed. For zero gravity options, piloted vehicles use opposition-class trajectories and are augmented by cargo vehicles using minimum-energy trajectories. The three mission cases are manifested as shown in tables 2.1.2-VII, -VIII, and -IX for the respective launch dates.

Lunar Observatory (Case Study 3). The major objective of the Lunar Observatory mission is the establishment of a major extraterrestrial science outpost. Emphasis will be placed on designing facilities for human deployment and operation of surface science systems. A LEO staging node will be required for onorbit assembly and checkout of spacecraft and transportation elements. Other characteristics of the missions are shown in table 3.1.2-X.

The manifest for the mission is shown in table 3.1.2-XI. From these heavy lift requirements, a 91-t HLLV capability will be required early in a Lunar Observatory program.

TABLE 3.1.2.-VII.- ETO MANIFEST — HLLV LAUNCHES FOR MARS MISSION 1

Launch	Date	Item	Mass (t)
Mars Cargo Vehicle (MCV) Components Launches for Assembly in LEO			
1. HLLV-1	Sep '04	MDV, TMIS engine	57.1
2. HLLV-2	Sep '04	TEIS, flight tanks	15.6
3. HLLV-3	Oct '04	Aerobrake, MOO, MTV,payload	91.0
4. HLLV-4	Nov '04	TEIS fuel #1, PhEV	66.4
5. HLLV-5	Nov '04	TEIS tank #2	58.7
6. HLLV-6	Dec '04	TMIS tank #1	
thru	thru	thru	
12. HLLV-12	Apr '05	TMIS tank #7 Launch Sep 2005	91.0 each
Mars Spaceship (MSS) Components for Assembly in LEO			
13. HLLV-13	May '05	Aerobrake, ECCV, TMIS engine	43.6
14. HLLV-14	Jun '05	3 SS & disk modules, supplies	68.1
15. HLLV-15	Jul '05	DSM, MOC, MOO	76.7
16. HLLV-16	Jul '05	DSM, MOC, MOO	76.7
17. HLLV-17	Aug '05	DSM, MOC, MOO	76.7
18. HLLV-18	Dec '05	TMIS tank #1	91.0
thru	thru	thru	
35. HLLV-35	Aug '06	TMIS tank #18 Launch Dec 2006	91.0

TABLE 3.1.2.-VIIa.- ETO MANIFEST — STS LAUNCHES FOR MARS MISSION 1

Launch	Date	Item
1. STS-1	Dec '04	Crew for teleoperated assembly of cargo vehicle
2. STS-2	Apr '05	Final check and crew return
3. STS-3	Jul '05	Teleoperated assembly of TMIS
4. STS-4	Nov '05	Crew rotation
5. STS-5	Mar '06	Crew rotation
6. STS-6	Jul '06	Crew rotation
7. STS-7	Dec '06	Flightcrew delivery

Lunar Outpost to Early Mars Evolution (Case Study 4). The major objective of this case is to provide an evolutionary development of a permanent human presence beyond LEO by supporting initial developments of a Mars outpost with a lunar outpost. Emphasis is placed on

TABLE 3.1.2.-VIII.- ETO MANIFEST — HLLV LAUNCHES FOR MARS MISSION 2

	Launch	Date	Item	Mass (t)
			Mars Cargo Vehicle (MCV) Components Launches for Assembly in LEO	
1.	HLLV-1	Sep '06	MDV, TMIS engine	57.3
2.	HLLV-2	Oct '06	3 TEIS flight tanks	10.5
3.	HLLV-3	Nov '06	Aerobrake, MOCS	40.8
4.	HLLV-4	Nov '06	MOO	89.4
5.	HLLV-5	Dec '06	TEIS fuel #1 DEV	77.5
6.	HLLV-6	Jan '07	TMIS tank #2 PhEV	79.6
7.	HLLV-7	Jan '07	TEIM fuel #3	69.3
8.	HLLV-8	Feb '07	TMIS tank #1	91.0
	thru	thru	thru	
16.	HLLV-16	Jul '07	TMIS tank #9 Launch Sept 2007	91.0 each
			Mars Spaceship (MSS) Components for Assembly in LEO	
17.	HLLV-17	Nov '07	Aerobrake, ECCV, TMIS engine	47.1
18.	HLLV-18	Dec '07	3 SS & disk modules, supplies	68.1
19.	HLLV-19	Jan '08	DSM, MOC, MOO	83.2
20.	HLLV-20	Jan '08	DSM, MOC, MOO	83.2
21.	HLLV-21	Feb '08	DSM, MOC, MOO	83.2
22.	HLLV-22	Mar '08	TMIS tank #1	91.0
	thru	thru	thru	
36.	HLLV-36	Dec '08	TMIS unit #15 Launch Feb 2009	91.0 each

TABLE 3.1.2.-IX.- ETO MANIFEST — HLLV LAUNCHES FOR MARS MISSION 3

	Launch	Date	Item	Mass (t)
			Mars Cargo Vehicle (MCV) Components Launches for Assembly in LEO	
1.	HLLV-1	Jan '09	MDV, TMIS engine	57.2
2.	HLLV-2	Feb '09	3 TEIS flight tanks	10.5
3.	HLLV-3	Mar '09	Aerobrake, MOO, MTV, payload	71.2
4.	HLLV-4	Apr '09	TEIS fuel	72.5
5.	HLLV-5	May '09	TMIS tank# 1	91.0
6.	HLLV-6	Jun '09	TMIS tank #2	91.0
7.	HLLV-7	Jul '09	TMIS tank #3	91.0
8.	HLLV-8	Aug '09	TMIS tank #4	91.0
9.	HLLV-9	Sep '09	TMIS tank #5 Launch Oct 2009	91.0
			Mars Spaceship (MSS) Components for Assembly in LEO	
10.	HLLV-10	Nov '09	Areobrake, ECCV, MIS engine	41.7
11.	HLLV-11	Dec '09	3 SS modules, disk module, supplies	68.1
12.	HLLV-12	Jan '10	DSM, MOO, MOC	75.0
13.	HLLV-13	Feb '10	DSM, MOO, MOC	75.0
14.	HLLV-14	Mar '10	DSM, MOO, MOC	75.0
15.	HLLV-15	Apr '10	TMIS tank #1	91.0
	thru	thru	thru	
24.	HLLV-24	Dec '10	TMIS tank #10 Launch Jan 2011	91.0

TABLE 3.1.2.-VIIIa.- ETO MANIFEST — STS LAUNCHES FOR MARS MISSION 2

Launch	Date	Item
1. STS-1	Jan '07	Crew for teleoperated assembly TEIS
2. STS-2	May '07	Crew rotation
3. STS-3	Sep '07	Final check and crew return
4. STS-4	Jan '08	Teleoperated assembly of TMIS
5. STS-5	May '08	Crew rotation
6. STS-6	Sep '08	Crew rotation
7. STS-7	Jan '09	Flightcrew delivery

TABLE 3.1.2.-IXa.- ETO MANIFEST — STS LAUNCHES FOR MARS MISSION 3

Launch	Date	Item
1.STS-1	Apr'09	Assembly and checkout of cargo vehicle
2. STS-2	Jul'09	Final assembly and checkout crew rotation
3. STS-3	Jan'10	Teleoperated assembly of TMIS
4. STS-4	Jun'10	Crew rotation
5. STS-5	Dec'10	Flightcrew delivery

TABLE 3.1.2.-X.- LUNAR OBSERVATORY — MISSION DESCRIPTION

Case Study Descriptor	Lunar Observatory
• Transportation	
- Trajectory profile	Free-return
- Number of flights	2 cargo, 2 crew (setup); 1 crew flt per year thereafter
• Crew size	4
• Crew total sortie time	≤ 20 days
• Surface stay time	< 14 days on surface
• EVA's (per mission)	12 six-hr EVA's (10-km unpressurized rover traverses)
• Propellant production	N/A
• Mission dates	
- Cargo	'04 & '05
- Piloted	'04 & '05 '06, '07, '08, '09, & '10 and continuing

TABLE 3.1.2-XI. ETO MANIFEST — HLLV LAUNCHES FOR LUNAR OBSERVATORY

Launch	Mission	Date	Item	Mass (t)
1. HLLV-1	* Car-1	Mar '04	LTV-1 (*19.0 t) LDV-C-1 (W) LDV-P-1 (D)	90.9
2. HLLV-2	Car-1	Apr '04	Propellant LTV-2 (D)	76.4
3. HLLV-3	** Hum-1	Jun '04	Propellant	79.0
4. HLLV-4	Car-2	Feb '05	LTV-3 (*19.0 t) LDV-C-2 (W) LDV-P-2 (D)	90.0
5. HLLV-5	Car-2	Mar '05	Propellant LTV-4 (D)	76.4
6. HLLV-6	Hum-2	Jun '05	Propellant	79.0
7. HLLV-7	Hum-3	Mar '06	LDV-P-3 (W) LTV-5 (D)	49.3
8. HLLV-8	Hum-3	Apr '06	Propellant	79.0
9. HLLV-9	Hum-4	Mar '07	LDV-P-4 (W) LTV-6 (D)	49.3
10. HLLV-10	Hum-4	Apr '07	Propellant	79.0
11. HLLV-11	Hum-5	Mar '08	LDV-P-5 (W) LTV-7 (D)	49.3
12. HLLV-12	Hum-5	Apr '08	Propellant	79.0
13. HLLV-13	Hum-6	Mar '09	LDV-P-6 (W) LTV-8 (D)	49.3
14. HLLV-14	Hum-6	Apr '09	Propellant	79.0
15. HLLV-15	Hum-7	Mar '10	LDV-P-7 (W) LTV-9 (D)	49.3
16. HLLV-16	Hum-7	Apr '10	Propellant	79.0
17. HLLV-17	Hum-8	Mar '11	LDV-P-8 (W) LTV-10 (D)	79.0
18. HLLV-18	Hum-8	Apr '11	Propellant	79.0

*CARGO **HUMAN

TABLE 3.1.2.-XIa.- ETO MANIFEST — STS LAUNCHES FOR LUNAR OBSERVATORY

Launch	Date	Item
1. STS-1	Mar '04	Teleoperated assembly (Car-1)
2. STS-2	Jun '04	Flight crew (Hum-1)
3. STS-3	Feb '05	Teleoperated assembly (Car-2)
4. STS-4	Jun '05	Flight crew (Hum-2)
5. STS-5	Mar '06	Flight crew (Hum-3)
6. STS-6	Mar '07	Assembly and flight crew (Hum-4)
7. STS-7	Mar '08	Assembly and flight crew (Hum-5)
8. STS-8	Mar '09	Assembly and flight crew (Hum-6)
9. STS-9	Mar '10	Assembly and flight crew (Hum-7)
10. STS-10	Mar '11	Assembly and flight crew (Hum-8)

establishing a robust infrastructure to maximize support to follow-on initiatives. First mission launch dates to the Moon are in 2004 and to Mars in 2010. Significant infrastructure buildup begins in 2001. Other case study characteristics are described in table 3.1.2-XII.

A manifest has not been established for this case because selection of implementation options is still fluctuating significantly. A few general characteristics of the ETO fleet requirements can be anticipated from the constraints established as part of the ground rules of this case. A specific mass-to-LEO budget has been established to drive the definition of infrastructure and transportation systems. This case will involve a significant number of lunar and Mars missions, probably involving some mix of chemical and electrically powered interplanetary vehicles, utilize significant orbit and surface infrastructure, and include the production, transport, and use of extraterrestrially produced propellants. A significant amount of ETO traffic will be required.

The maximum payload capacity requirements for the HLLV will likely ramp-up at a pace similar to that in the other case studies, and by 2010 a 91- to 200-t-payload-capacity HLLV will probably be required. If nuclear electric propulsion is chosen, most of the manifest will be comprised of hardware rather than propellants, as in the other studies. An ETO development program for HLLV's with large payload capacities and a high degree of reusability will be required.

3.1.3 Prerequisite Program Accommodation of Case Studies

3.1.3.1 ETO Capability Growth with Time

Current plans are for the ETO transportation capability to grow via increases in the STS lift capability; the addition of an unmanned STS cargo element, Shuttle-C; and

TABLE 3.1.2.-XII.- LUNAR OUTPOST TO EARLY MARS EVOLUTION — MISSION DESCRIPTION

Case Study Descriptor	Lunar Outpost to Early Mars Evolution	
	Lunar Portion	Mars Portion
• Transportation		
- Trajectory profile	Cargo: low thrust Crew: free-return translunar	Cargo: low thrust Crew: near fuel minimum
- Number of flights	Continuing LEO/LLO LLO/LS Shuttle	1 cargo, 3 crew
• Crew size	8	8 (8 to Mars surface)
• Crew total sortie time	< 1 Year	35 to 45 mos.
• Surface staytime	< 1 Year	2 Years
• EVA's (per mission)	(10-km unpress rover traverses)	10-km unpress rover traverses and 100-km press rover traverses
• Mass to LEO peak year	Peak 345 t @ 2005	
• Max propellant mass (t)	Total: 1660 t (all flights, lunar and Mars)	
• User allocation (t)		
- Orbital	3.3 t	12 t
- Surface	112 t	Mars 58 t Deimos 10 t
• Propellant production	Four LOX plants (40 t each)	Phobos prop. plant (86 t)

the development of an HLLV, the advanced launch system.

A second-generation manned vehicle and options for additional upgrades to the current STS are also being studied. Both of these studies include options for additional personnel-carrying capability that may be required to support the human exploration missions. A concept for a second-generation manned vehicle, with a passenger carrier replacing the usual payload canister, is shown in figure 3.1.3-1.

Projected growth in the mass-per-launch capability is shown in figure 3.1.3-2. A scenario of the evolutionary HLLV capabilities in figure 3.1.3-3 shows a development cycle that starts with the common core vehicle, continues with an augmentation to 68 t with strap-on solid boosters, and ultimately evolves to a 91-t vehicle through addition of cryogenic/hydrocarbon liquid rocket boosters. An additional evolutionary step may be the use of a flyback booster to achieve lower recurring costs.

3.1.3.2 Schedules

ETO transportation for all case study missions can be provided by vehicles currently in OSF planning, as shown by figure 3.1.3-4. However, accomplishing the Phobos expedition with one or two launches will require a very large vehicle (magnum), which is not in current OSF plans. Total launch mass required for an all-propulsive Phobos mission would be 1778 t (3911 klbs); for an aeroassist mission, 765 t (1683 klbs).

3.1.4 Support Required from Other NASA Organizations

3.1.4.1 Headquarters Program Coordination

Close coordination with other program offices is required in several areas. OSF and the Office of Aeronautics and Space Technology (OAST) are coordinating efforts to ensure that promising transportation system technologies are efficiently transferred to the OSF advanced development program. OSF is coordinating with the Office of Space Station (0SS) and OAST on the technology, facilities, and structures for an orbital transportation node to support refueling and maintenance of space-based vehicles. OSF is coordinating with the Office of Space Science and Applications (OSSA), as well as other transportation system users, to ensure that the transportation system will service automated/robotic missions as well as human exploration. OSF works with users to collect summaries of all mission needs, which are updated annually and published as the Civil Needs Data Base, the basic mechanism for ensuring that the total civil needs are reflected in transportation planning.

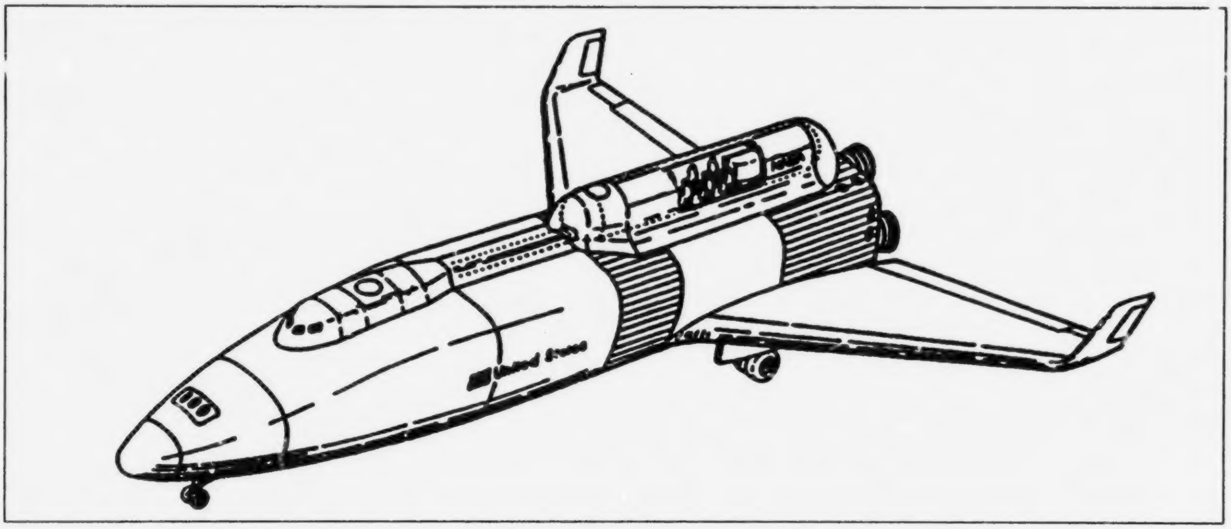


Figure 3.1.3-1.- Passenger/logistics carrier integrated into second-generation manned vehicle.

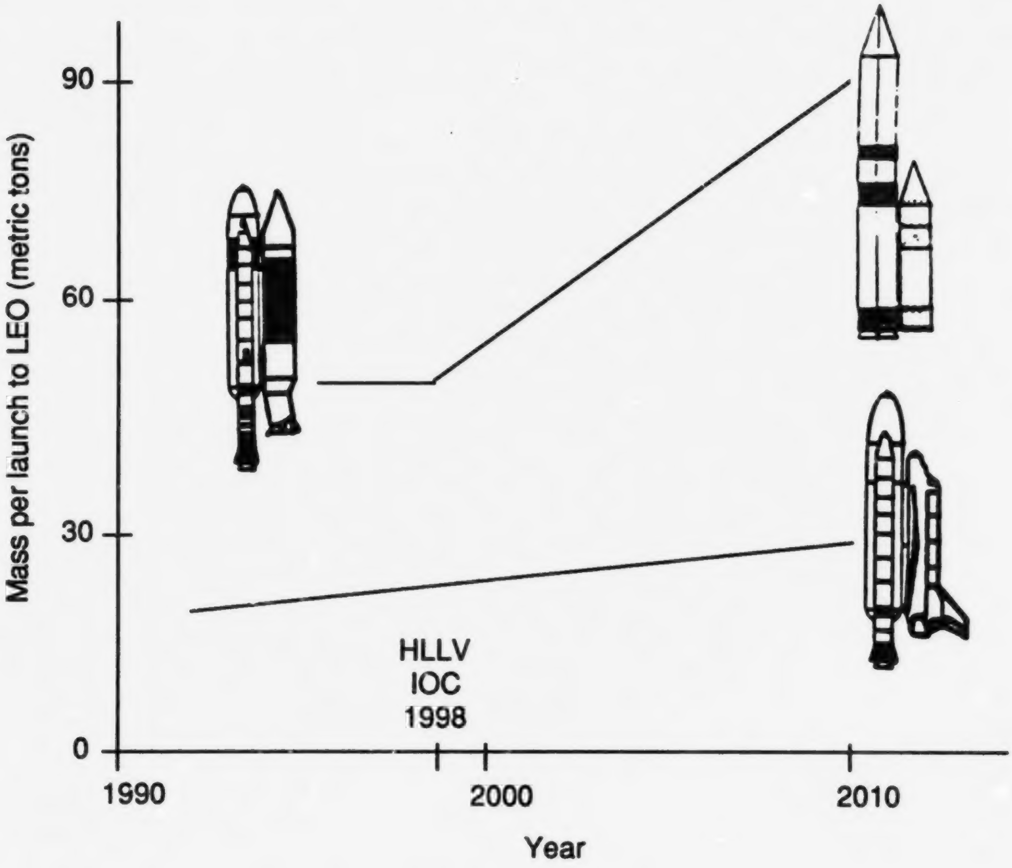


Figure 3.1.3-2.- ETO capability growth with time.

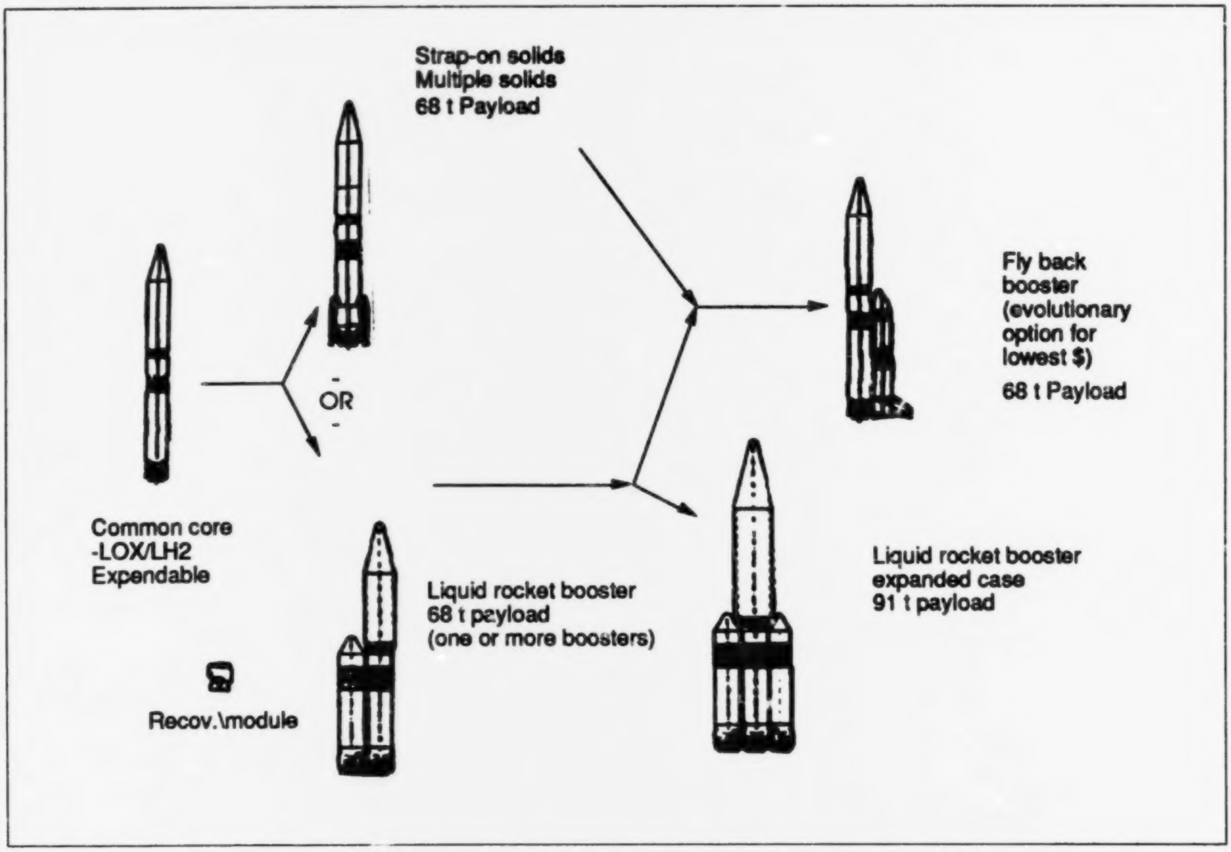
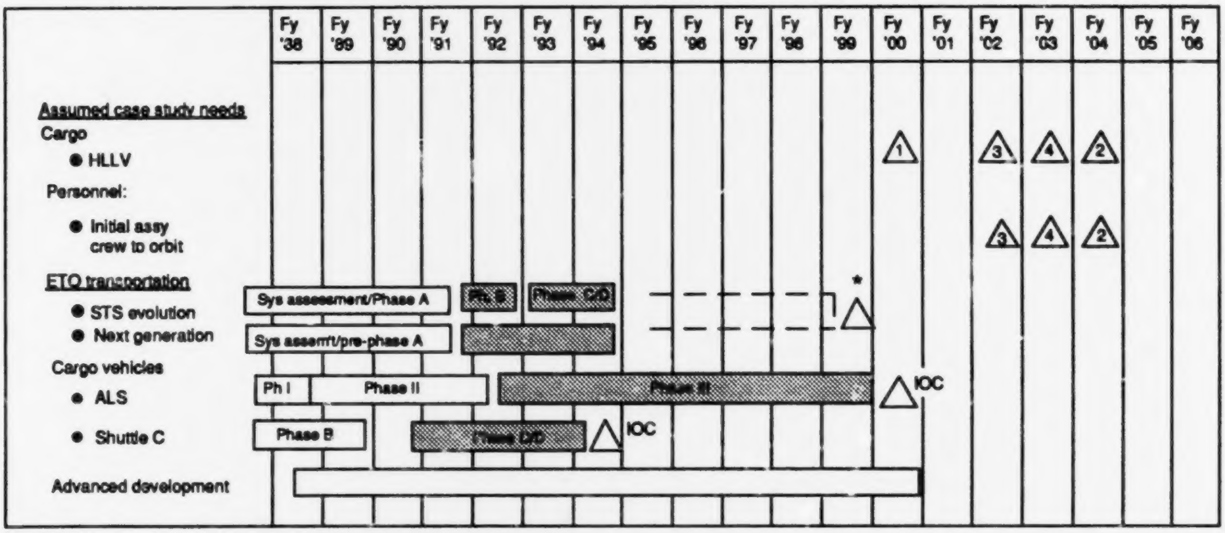

Strap-on solids
Multiple solids
68 t Payload
Fly back
booster
(evolutionary
option for
lowest $)
68 t Payload
OR
Common core
-LOX/LH2
Expendable
Liquid rocket booster
68 t payload
(one or more boosters)
Liquid rocket booster
expanded case
91 t payload
Recov.\module

Figure 3.1.3-3.- Evolution of HLLV capabilities.



* IOC depends on requirements & concepts selected

X x=Case study number need date

IOC=Initial operational capability

Figure 3.1.3-4.- Advanced transportation program (FY 1989 planning).

3.1.4.2 Major Technology Areas

A propulsion-focused advanced technology development plan which details a coordinated series of tasks is underway. One of the key technology areas is the liquid propulsion system composed of the space transportation booster engine (STBE) and the space transportation main engine (STME). The STBE can provide main boost propulsion for the nation's next generation HLLV. To support the engine definition activities, a baseline gas-generator-cycle engine configuration is being used. This engine falls within the performance spread currently being evaluated for the HLLV. The STBE development schedule will support an HLLV launch in the mid- to late 1990's. The current baseline version of this high-performance liquid rocket engine system will use liquid oxygen and hydrocarbon/methane fuel. The STBE will be designed for low cost, high reliability, and long life with very low maintenance. The current baseline STBE primary performance characteristics are

a. Thrust - 625 klb (sea level)
b. ISP - 325 s (sea level)
c. Chamber pressure - 3000 psi
d. Area ratio - 48:1
e. Engine weight - 6500 to 7500 lbs.

The STME can provide ETO transportation propulsion for the next generation of launch vehicles. The STME will be a robust, simple, inexpensive expendable rocket engine. Low-cost design philosophy is being maintained throughout the activities to ensure that the cost goals are achieved. STME performance characteristics are

a. Thrust - 435 klb (vacuum)
b. SP - 447 s (vacuum)
c. Chamber pressure - 2400 psi
d. Area ratio - 75:1
e. Engine weight - 6500 to 7500 lbs

Focused technology and new development activities are divided into five major areas:

a. LOX/hydrogen engine
b. LOX/hydrocarbon engine
c. Booster/core propulsion subsystems
d. Solid propulsion
e. Facilities

The primary emphasis of these activities is the development and demonstration of design concepts, manufacturing processes, and techniques that will provide significant cost reduction over current systems.

3.1.5 Summary

3.1.5.1 Major Trade Studies

In analysis of the ETO capability requirements, three major trade areas were identified for the next study cycle:

a. HLLV capability versus onorbit assembly
b. Propellant transfer
c. Reusable HLLV's versus expendables

In the next cycle of the study, further definition and quantification of the relation between complexity of orbital operations and HLLV capability should be determined and used in related analyses. In the coming months, refinements of space transfer vehicle concepts will allow the trade between mass and volume capabilities and onorbit assembly to be conducted.

Propellant handling in zero gravity has been the subject of considerable small-scale research, but with the volume and mass needed for the cases studied, additional study and technology development will be necessary. The time required to pump 91 t of cryogenic propellant from an orbital tanker into an orbital tank facility and later transfer it to a space transfer vehicle may be excessive. If the space transfer vehicle tanks can be designed for launch from Earth fully or partially filled, the time could be minimized. However, special facilities will be required onorbit for docking with the cryogenic tanks.

The high cost of space hardware has resulted in consideration of reusable ETO elements in almost every new program. In many instances, such as the Apollo program, the limited number of units involved favored the expendable option. In Case Studies 2 and 4, the launch rates did indicate some degree of reusability. Due to additional systems requirements for reusability, ETO-mass capability is reduced. In Case Studies 1 and 3, it is not apparent that reusability is advantageous if no further missions are planned beyond the ones designed in the case studies. Refurbished costs need to be quantified and loss of performance determined.

3.1.5.2 Concerns and Issues

Reliability of HLLV for Multilaunch Missions. For missions that require a relatively large number of HLLV launches, the reliability can be a crucial factor in mission design since the loss of a single flight could jeopardize the entire mission. The sparing of critical in-space elements could be expensive, time-consuming, and a significant cost factor. Reliability specifications for the HLLV must be addressed in this context.

Onorbit Assembly. Some onorbit assembly, of complex mating and deployment operations, will be required for these missions. A basic question of NASA's approach to future missions operations is involved. Does the expansion of human presence into the solar system require development of in-space operational capabilities (starting in LEO) or can in-space assembly be minimized by making the launch vehicle and the transfer vehicle large enough to transport assembled elements? The approach that is chosen can drive many aspects of future missions.

People Traffic to LEO. The mission manifests have been summarized here in terms of total mass to LEO, with the people traffic accommodated on the STS flights. Decisions such as the degree of onorbit assembly as well as the mission profile will affect the requirement for people traffic to LEO. In the context of total national needs, additional personnel-carrying capability may be required.

Ground Processing Facilities. The requirements for ground processing facilities were not addressed in depth in this study cycle. The importance of this aspect of the missions can be appreciated by examining the launch rates and the size of the vehicles required. These factors are interdependent with the questions of onorbit assembly and propellant transfer capabilities. Much larger propellant handling facilities will be required in any case. Extensive examination of all of these factors as well as the development of more detailed concepts for the ground launch facilities will be required in the next study cycle.

Propellant Storage and Handling. Propellant is the major component of the mass-to-orbit requirements; therefore, the method of transporting and transferring it must be examined carefully. The issue is whether the unmanned cargo vehicle is configured as a tanker, or propellant is prepackaged in small tanks for transfer directly to the space transfer vehicle. This is a function of the node infrastructure, but the transfer method also impacts the mass-to-volume ratio of the HLLV.

Magnum Launch Vehicle. The concept of transporting the complete Phobos mission system to orbit in one launch has the merit of possibly eliminating orbital operations and limiting the ground launches to one. However, neither the vehicle nor the ground operations for this has been studied. A concept that seems attractive for its simplicity may prove to be much less than simple in practice. Additionally, the issue of risking the complete mission hardware on a one launch for which there is little or no experience base must be faced. Even if the Phobos space transfer vehicle is launched dry—to be fueled onorbit, this issue remains, as does the problem of fueling the vehicle with no support from an orbital assembly node. This concept also raises the question of whether an infrastructure is being developed to support a single mission or to support continued expansion of the human presence in the solar system.

3.1.5.3 Comments

Planning the transportation to start humans on the path to exploring the solar system is an exciting challenge. Although the first destination has not been chosen and the first launch date seems far away, today's decisions and actions with respect to the transportation system will affect the capability to achieve the goals that are chosen in 1992. Evolutionary capabilities that are built into the design of transportation elements will make the transition to the human exploration era easier and more cost-effective as will the investments in studies and technologies. Similarly, decisions on the method of implementation of this human exploration mission will affect the options for going even further in the future. The trades that are conducted in these studies must consider the entire goal of expanding the human presence into the solar system, not just the first step.

3.2 EXPLORATION IMPACTS TO UNMANNED SOLAR SYSTEM MISSION PROGRAMS

3.2.1 Role of the Office of Space Science and Applications (OSSA) Solar System Exploration Division in Manned Exploration

The current unmanned solar system exploration program, as administered by OSSA (code EL), represents an ambitious and comprehensive effort through both scientific research and a series of coordinated, unmanned flight missions designed to answer fundamental questions about the origin and evolution of the solar system. Many of these unmanned flight missions will serve as precursors to manned expeditions by focusing the development of required technologies and demonstrating the engineering capabilities needed to safely and productively conduct a program of manned exploration.

The blending and cooperation of the unmanned exploration program with one of human exploration provides a very natural, synergistic, and economic application and extension of the unmanned exploration program to the national goal of human exploration of the space near Earth.

3.2.2 Case Study Needs Assessment

3.2.2.1 Precursor Mission Set

As applied to the study of the Mars and lunar systems and to national human exploration goals, the current unmanned exploration program now consists of three planned flight programs in varying stages of maturity:

Mars Observer (MO), Mars Rover Sample Return (MRSR), and Lunar Observer (LO). Other predominantly scientific missions with the potential to provide significant precursor information, such as a Phobos probe (or possibly a sample return) using a Mariner Mark II (MM II) spacecraft, are now being evaluated.

All of these missions would be managed by OSSA and are being planned according to the schedule in table 3.2.2-I.

These missions now constitute the precursor mission set (PMS), and when integrated into the overall strategy for human exploration, they will play a key role in leading the way for the successful implementation of a human exploration program. The PMS will satisfy as many of the precursor requirements as practical, provide as many technology and/or engineering demonstrations as possible, and expand the knowledge base. These missions will also reduce the safety risks to the flightcrews who participate and will advance the engineering and technology essential to the design of the spacecraft and systems they will fly.

3.2.2.2 Precursor Mission Support Assessment

An assessment of the overall precursor missions support to the four exploration case studies indicates that the MRSR and LO precursor missions are in the exploration-critical path. Ideally, the results of science precursor requirements and technology demonstrations should be available at or near the start of the explorations program to influence the design of spacecraft and systems.

Table 3.2.2-II compares desired dates for precursor mission data to dates the data could be expected to support the case study milestones.

TABLE 3.2.2-I.- PRECURSOR MISSION SET

Precursor Mission	Schedule
Mars Observer	1992 Launch
Mars Rover Sample Return	1992 or 1993 Start
Lunar Observer	1992 or 1993 Start
Other unmanned, predominantly science-oriented missions	(TBD)

TABLE 3.2.2-II.- ASSESSMENT OF PRECURSOR MISSION SUPPORT TO HUMAN EXPLORATION PROGRAM

	Exploration Case Study			
	Phobos	Mars	Lunar Obs	Evol.
• Desired date for precursor requirement completion	1996	1997	1998	1996-1999
• Earliest available precursor mission date to satisfy requirement				
• Mars Observer	1993	1993	N/A	1993
• MRSR	N/A	1998-2002	N/A	1998-2002
• Lunar Observer	N/A	N/A	1997	1997
• Phobos (USSR)	1989	1989	N/A	1989

Figure 3.2.2-1 plots the availability of science precursor mission results and technology demonstrations from Phobos (USSR), MO, and a MM II probe to Phobos within case study milestones. The results of the USSR Phobos and MO missions would be timely to the design of the Phobos cargo and piloted elements. However, an MM II launch in late 1998—as a precursor to a manned Phobos mission—would provide useful data in 1999, but would not provide timely information until very late in the phase C/D for the piloted and cargo vehicle design.

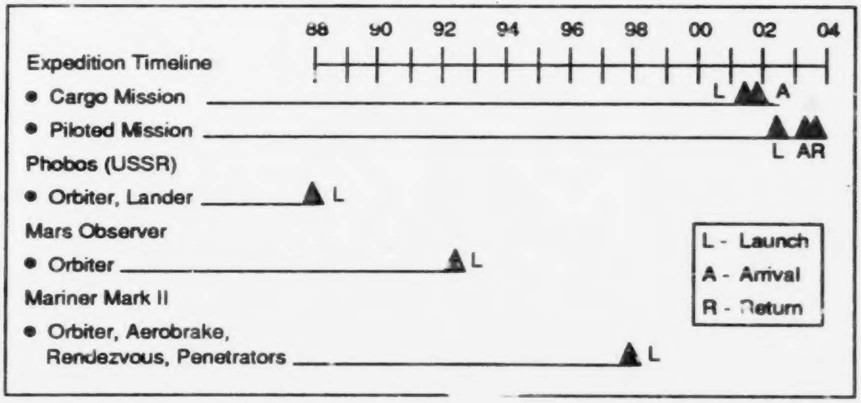


Figure 3.2.2-1.- Precursor mission compatibility assessment for the Phobos case study.

Similarly, figure 3.2.2-2 shows the criticality of the MRSR schedules to support the Mars and Evolutionary case studies. For the MO and MRSR precursors to the Mars case study, many of the mission results would be available. Key technology demonstration items such as Mars aerocapture would be accomplished on the Orbiter/network launch of MRSR option 1. However, a sample from Mars would not be returned to Earth until late in phase C/D of the Mars cargo and piloted programs.

3.2.3 Prerequisite Program Accommodation of Case Studies

3.2.3.1 Precursor Science-Related Requirements

The exploration scenario development will lead to the definition of operations, systems, information (science), and technology demonstration requirements which must be satisfied prior to manned operations. As these precursor requirements are developed in increasing specificity and detail, they will be assessed against the current scientific/ technical knowledge base and the precursor mission set. The current science-related precursor requirements are addressed by the precursor missions as shown in table 3.2.3-I.

3.2.3.2 Engineering Tests and Demonstrations

In addition to specific science requirements definition, OSSA will also participate in some generalized engineering tests and/or demonstrations such as that of aerocapture or autonomous rendezvous and docking. These demonstrations are now considered part of the preliminary set of precursor requirements and could be conducted on one of the appropriate precursor missions. A current assessment of the tests and demonstrations is shown in table 3.2.3-II.

TABLE 3.2.3-I.- ASSESSMENT OF SCIENCE REQUIREMENTS ADDRESSED BY PRECURSOR MISSIONS

Precursor Science Requirements	Precursor Mission		
	MO	MRSR	LO
• Solar flare detection			
• Surface material properties		X	
• Use of indigenous materials	X	X	X
• Surface characteristics	X	X	X
• Radiation shielding		X	X
• Natural hazards	X	X	X
• Martian moon utilization			
• Transportation hazards	X	X	
• Contamination issues		X	

TABLE 3.2.3-II.- ASSESSMENT OF TECHNOLOGY & ENGINEERING DEMONSTRATIONS ON PRECURSOR MISSIONS

Technology and engineering demonstrations	Precursor missions			
		MRSR		
	MO	Option 1	Option 2	LO
• Landing accuracy		'00	'00	
• Mars aerocapture		'98	'00	
• Earth aerocapture		'02	'04	
• Surface mobility		'00	'00	
• Rover power		'00	'00	
• Auto rend. & docking		'01	'03	
• Sample acquisition & handling		'00	'00	
• Sample return		'02	'04	
• Rover operations		'00	'00	

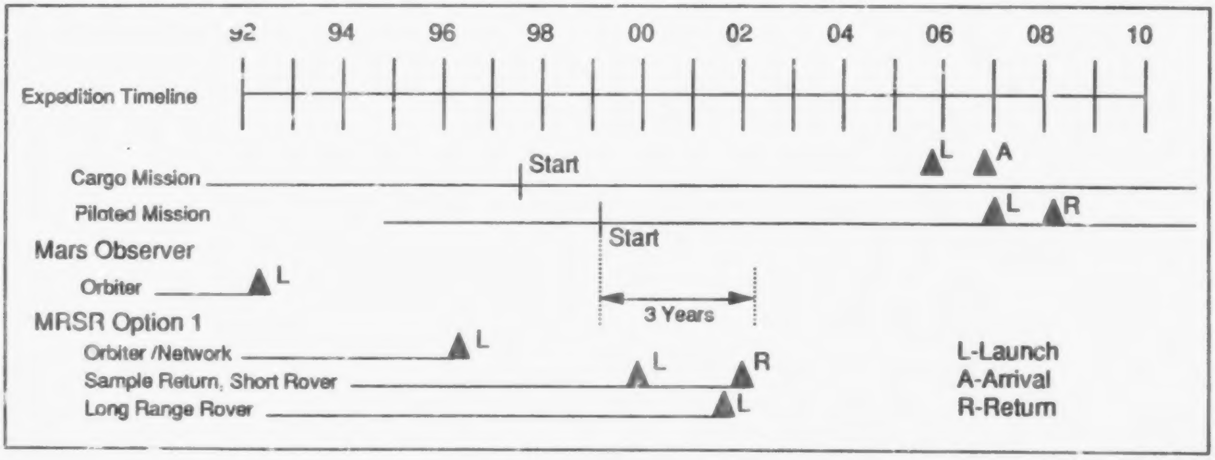


Figure 3.2.2-2.- Precursor mission compatibility assessment for the Mars case study.

3.2.3.3 Analysis of Precursor Requirements

OSSA will continue to analyze precursor requirements and provide recommendations to satisfy them. Each of the precursor requirements must be refined and validated, and plans must be made to satisfy each. A group (or groups) with the necessary scientific and engineering experience must be chartered to produce this refined set.

In addition to assessing the precursor requirements, this group would be responsible for studying and recommending the extent to which the precursor mission set could, naturally or through modifications, contribute to satisfying the precursor requirements. A possible outgrowth of this activity would be the study of and/or recommendations for additional precursor mission(s).

Table 3.2.3-III provides an initial assessment of some of the precursor requirements which will not be completely addressed by the current or planned precursor missions. Note that the Solar System Exploration Division (SSED) has no plans that will address the requirements for solar flare detection and prediction.

3.2.3.4 MRSR Timelines

A number of MRSR mission options are currently being studied, two of which are discussed herein. Option 1 provides for a short-range rover to obtain an early sample and, later, for a long-range rover for follow-on science studies. Option 1 results in a sample being returned to Earth in 2002. A schedule for the more significant milestones is shown in figure 3.2.3-1.

A second option, for the start of development of a long-range rover concurrent with an orbiter in 1994 followed by a sample return element in 1996, would result in a sample to Earth in the year 2002 versus 2004. As can be seen in figure 3.2.3-2 for option 2, this sample return is too late to satisfy the precursor requirement; however, the technology and engineering demonstration of aerocapture and accurate landing and many other demonstrations could be achieved.

To provide MRSR precursor results on a schedule more compatible with the Mars exploration case studies, consideration was given to compressing the MRSR time-

TABLE 3.2.3-III.- INCOMPLETELY COVERED PRECURSOR MISSION REQUIREMENTS

Requirement	Precursor Missions				
	MO	MRSR Option 1	MRSR Option 2	LO	Phobos (USSR)
• Site selection	Partial	Partial	Partial	Partial	Partial
- Candidate site list					
- Site survey					
- Site certification					
• Solar flare detection					
• Martian moon utilization					Partial
• Contamination issues	Partial	Partial	Partial		

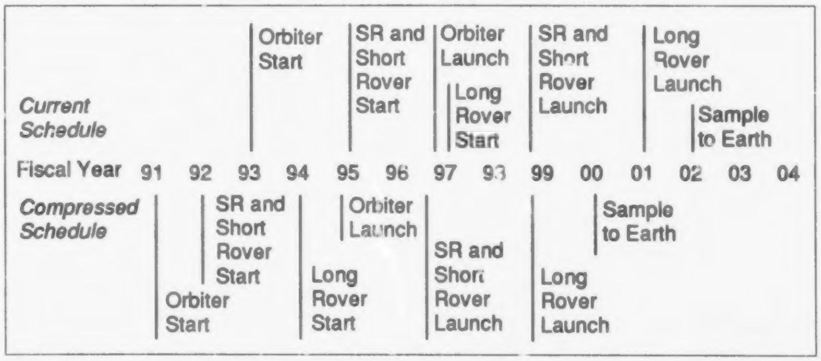

Current Schedule
Orbiter Start
SR and Short Rover Start
Orbiter Launch
Long Rover Start
SR and Short Rover Launch
Long Rover Launch
Sample to Earth
Fiscal Year
91
92
93
94
95
96
97
98
99
00
01
02
03
04
Compressed Schedule
Orbiter Start
SR and Short Rover Start
Long Rover Start
Orbiter Launch
SR and Short Rover Launch
Long Rover Launch
Sample to Earth

Figure 3.2.3-1.- MRSR option 1 current and compressed timelines.

lines; the resulting schedules are shown in the lower half of figures 3.2.3-1 and -2 for options 1 and 2 respectively. These compressed schedules, although more compatible with the exploration case studies, require a significant funding increase from the current levels both in 1989 and 1990. The compressed MRSR timelines were discarded because of the anticipated higher funding requirements.

3.2.3.5 MRSR Launch Vehicle Requirements

The MRSR launch vehicle support requirements for options 1 and 2 for both the current and compressed schedules are given in table 3.2.3-IV.

3.2.4 Support Required from Other NASA Organizations

A high degree of interaction and cooperation of the various Headquarters program offices in defining, assigning, and satisfying precursor requirements is essential to successful implementation of the human exploration program. The process to be evolved for effecting that interaction and cooperation will build naturally on the existing organizational structure and traditional roles and missions of the program offices. The SSED will continue to concentrate on the planetary science and engineering aspects.

3.2.5 Summary

The SSED has three currently planned unmanned missions which can be used to satisfy most of the currently defined human exploration science and engineering precursor requirements. If these missions prove sufficient, the impacts of human exploration on SSED should be relatively minor.

All legitimate precursor missions should be protected with sufficient funding to permit appropriate mission completion when the exploration decision is made.

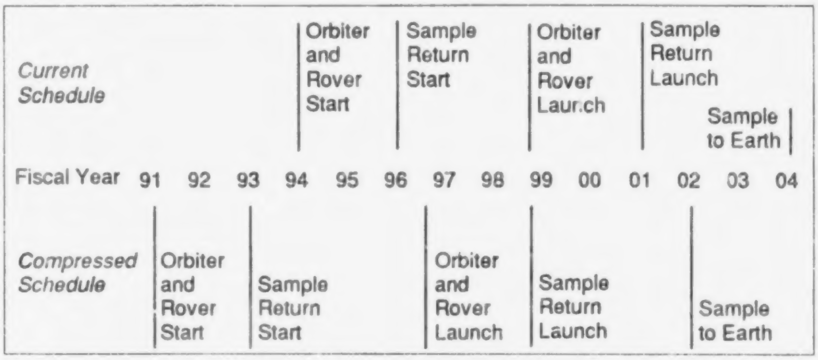

Current Schedule
Orbiter and Rover Start
Sample Return Start
Orbiter and Rover Launch
Sample Return Launch
Sample to Earth
Fiscal Year 91 92 93 94 95 96 97 98 99 00 01 02 03 04
Compressed Schedule
Orbiter and Rover Start
Sample Return Start
Orbiter and Rover Launch
Sample Return Launch
Sample to Earth

Figure 3.2.3-2.- MRSR option 2 current and compressed timelines.

TABLE 3.2.3-IV.- MRSR LAUNCH VEHICLE REQUIREMENTS

	Flight Package	Current	Compressed
OPTION 1	Orbiter/network	2 veh 10/96	2 veh 09/94
	Sample return Vehicle with short rover	2 veh 11/98	2 veh 10/96
	Long-range rover	2 veh 01/01	2 veh 11/98
OPTION 2	Orbiter/rover	2 veh 11/98	2 veh 10/96
	Sample return vehicle	2 veh 01/01	2 veh 11/98

3.3 EXPLORATION IMPACTS TO SPACE LIFE SCIENCES

3.3.1 Role of the Office of Space Science and Applications (OSSA) Life Sciences Division in Manned Exploration

The NASA Office of Exploration (OEXP) is presently examining the requirements for, and precursor activities implied by, four candidate missions: a human expedition to Phobos, human expeditions to Mars, human-tended lunar observatories, and lunar outpost to early Mars evolution. The HQ Life Sciences Division (code EB) pursues two important goals, both of which have direct relevance to such missions. The Basic Science goal has as its objective the use of the space environment to conduct basic science research that will increase our understanding of life sciences processes on Earth and expand our understanding of the origin and evolution of life in the universe. The relationship of this life sciences goal to exploration missions is that the advances it offers in human knowledge of the solar system constitute a reason for the conduct of such missions. A second and equally important goal of the Life Sciences Division is to specify the requirements and develop the technologies that must be in place to ensure the safety, health, and productivity of human participants of exploration missions. This is the Enabling goal of life sciences, so called because its objectives must be met before manned missions are undertaken. Since it is the purpose of this section to examine what must happen before exploration missions can proceed, the focus will be on the enabling activities of life sciences; i.e., on the science and technology necessary to support the candidate missions generally, and each of the four candidate missions considered individually.

Potential contributions to exploration by international partners and the USSR are not factored into this plan. However, such contributions can be substantial and the Life Sciences Division plans to pursue a vigorous international cooperation program which is already in place for spacelabs and Space Station Freedom.

3.3.2 Case Study Needs Assessment

3.3.2.1 Near-Term Plans of Enabling Life Sciences

Five life sciences areas have been identified as critical to supporting the conduct of exploration missions as described in the four candidate case studies. This section will describe ongoing and planned activities in these life sciences areas and relate them to case study needs.

Section 3.3.3 will discuss the compatibility of ongoing and planned life sciences programs with case study need dates. As will be shown, not only must these near-term planned programs be strongly supported, but two new programs need definition and consideration for incorporation into the planned programs. These are a variable-gravity research program and an advanced life support system for the Mars case and the Mars portion of the evolutionary case. The five critical life sciences areas are

a. Advanced medical care — the provision of remote medical care in the event of illness or injury

b. Artificial gravity/countermeasures — an appraisal of how to maintain the health and physical capabilities of crews during exposure to micro- or reduced gravity, and of how to facilitate readaptation to an Earth-gravity environment

c. Radiation — the determination of chronic low-dose and solar flare radiation risks and the development of appropriate countermeasures and warning capability

d. Life support — development of processes for revitalizing air and water, supplying food, and monitoring and decontaminating of the environment

e. Space human factors — optimizing systems design requirements and measures to ensure safe, productive, and enhanced crew performance

All five of these areas are either actively under development as part of the life sciences ongoing program or are planned for near-term investigation as the life sciences part of the OSSA Strategic Plan. Although all areas play some role in each of the case studies, the demands of the several areas vary with particular case studies. The variation relates both to the net importance of the area to that particular case study and to the question of how much information remains to be gathered before this area can be considered acceptably addressed for the case study conditions. The paragraphs that follow provide the basis for these assessments. Table 3.3.2-I presents broad general assessments of the importance of an area to the several case studies.

Advanced Medical Care. Portions of Health Maintenance Facility (HMF) presently under development will be flown and evaluated on early flights of spacelabs and the extended duration orbiter, becoming operational on Space Station Freedom. This facility will provide for onboard diagnostics, therapeutics, monitoring, countermeasures, and medical information management. An advanced version of the HMF is also planned as part of the advanced technology development program. These facilities are expected to meet near-term needs for the health care of astronauts. As applied to exploration missions, these facilities should prove adequate for the Lunar Observatory mission and the lunar portion of the Lunar Outpost to Early Mars Evolution mission. In both cases,

TABLE 3.3.2-I.- TECHNOLOGIES REQUIRED FOR MISSION CASE STUDIES

Technology/Mission	Phobos	Mars	Lunar Observatory	Moon-Mars Evolutionary	
				Phase 1	Phase 2
Adv. medical care	H	H	L	L	H
Reduced-g countermeasures	H	H	L	M	H
Space human factors	H	H	L	M	H
Life support processes	H	H	L	M	H
Radiation protection	H	H	M	H	H

H — Critical importance and/or low level of information
M — Moderate importance and/or some information
L — Low importance and/or acceptable level of information

the relative proximity of the astronauts to the space station and the possibility of quick rescue and a return to Earth lessen the need for more extensive onboard facilities. In the cases of the Human Expedition to Phobos mission, the Human Expeditions to Mars mission, and the Mars portion of the Lunar Outpost to Early Mars Evolution mission, rescue will not be possible and a high level of crew medical self-sufficiency will be a requirement. The relative autonomy of exploration mission astronauts dictates that health care capabilities be extended beyond those of Space Station Freedom. This means that onboard computer-aided diagnosis systems; automated, miniaturized clinical chemistry systems; and a general surgery capability will be necessary. An evaluation will be needed to determine the scope and design of an autonomous HMF and also to ascertain the skills required to make the operations of such a facility practical. Since space station technology is adequate for a lunar operation, a lunar outpost could be used to provide the reduced gravity environment in which to assess the requirements and develop the approaches necessary to meet the needs anticipated on a Phobos or Mars mission. In all mission scenarios, the capability to provide medical care will be defined both by medical support equipment and by the medical skill of the crew.

Zero-g Countermeasures/Artificial Gravity. One of the most pressing requirements of space flight, and especially of long-duration space flight, is to counteract the negative physical and physiological effects of microgravity. Space motion sickness, although a significant and even debilitating problem to some spacefarers, at least can be expected to subside after several days of exposure to weightlessness. However, the sustained, undesirable, and potentially health-threatening effects of weightlessness (e.g., bone loss and muscle loss) are long-standing, tend to be cumulative, and require that intervention techniques or countermeasures be employed. Just what combination of countermeasures and procedures is necessary or preferred under what flight conditions is not yet adequately understood. So far, both Americans and Soviets have relied heavily on exercise regimes, usually vigorous and protracted, to provide the desired protection. At the present time it is unclear whether exercise countermeasures will be capable of maintaining crew health for very long duration missions. It will be difficult for astronauts to keep up the required exercise program for the protracted periods of most exploration missions. If an astronaut should suffer an accident or a serious illness and be unable to exercise, more severe deconditioning could result. It is the plan of the Life Sciences Division to use the opportunities provided by Space Station Freedom to improve our understanding of the biomedical effects of weightlessness and particularly of the extended duration effects. The extended duration crew operations (EDCO) verification on Space Station Freedom will provide demonstrations of 6 months and longer crew exposures to microgravity. This extends by more than 3 months the longest previous American flight; i.e., the flight of Skylab 4. In addition to a better understanding of biomedical effects of prolonged weightlessness and the efficacy of exercise in maintaining conditioning, the Life Sciences Division program on Space Station Freedom will allow for the testing of alternate or supplementary countermeasures such as diet, pharmaceuticals, and elsewhere-mart stimulation. More importantly, it will allow initial assessments to be made with animal subjects of the health maintenance value of artificial gravity.

Artificial gravity has never been tested in space but has been proposed as a useful technique for managing the long-term effects of weightlessness. Artificial gravity alone is unlikely to offset bone loss, muscular atrophy, and changes in the cardiovascular system, and it may introduce vestibular complications. However, it is probable that a carefully-developed regimen of artificial gravity combined with exercise would provide the desired results. Gravity could be simulated in weightlessness either by rotating the entire spacecraft or by carrying a human-rated centrifuge along with or onboard the space vehicle. However, it is not certain if any of these centrifugation approaches will result in the same positive effects as gravity. In addition, unique costs are associated with providing artificial gravity; thus its use may be indicated only if other, less expensive, solutions are found to be inadequate. Artificial gravity as a countermeasure is not a requirement of short-duration missions and therefore is not part of the present Life Sciences Division's program. It is, however, an element in the Life Sciences Division's Strategic Plan and advanced technology studies. A long lead time is required to determine if artificial gravity is effective and needed, and the plan is to begin to explore this issue through the use of freeflying satellites, both as part of the basic Life Sciences Division's program and through the humans in-space (HiS) element of Pathfinder. Construction and flight testing on Space Station Freedom of the 1.8-m centrifuge is part of the artificial gravity research plan, and it is desirable to begin at once the definition studies on artificial gravity devices. Once an understanding of artificial gravity effects is secured, it may be deemed necessary to go to the next step; i.e., to explore the use of a simulated range of gravities between the 0 g of space and the 1 g of earth. If so, a variable gravity research facility (VGRF) would be constructed and tested. However, it must be noted that the OSSA Strategic Plan must be augmented if the VGRF becomes necessary,

The requirements for artificial gravity/countermeasures vary with the case study. For the sporadically tended lunar observatory, countermeasures found adequate for Shuttle and Space Station Freedom should suffice. An expedition to Phobos may require the use of artificial gravity. Similarly, an expedition to Mars or a Mars outpost may require artificial gravity, with the added complication that the effects of the 1/3 g of the Mars environment in the landing crew after a long 0-g stay are not known. For the lunar outpost portion of the evolutionary case study, it is possible that a long-term stay on the lunar surface may be accomplished with the 1/6 g of the lunar environment alone, or by a combination of lunar gravity and other countermeasures. Alternately, it may be found that an artificial gravity mode will also be required on the lunar outpost.

Radiation. Radiation threats in space are in the form of both chronic, low-dose cosmic radiation, and infrequent, high-dose solar flare events. The ongoing life sciences program has limited resources directed towards investigating and providing protections against space radiation. A basic need is to define the nature and biological effects of the space radiation environment. A freeflyer capability is presently planned, with phase C/D scheduled for 1991. Although space flight availability is limited and biological dosimetry measurements have been difficult to schedule, it is anticipated that the freeflyer polar mission could be launched as early as 1992. Such missions would provide a much better understanding than presently available of the biological radiation risks of space flight. Extension of the radiation work is also planned as part of the HiS element of Pathfinder.

A solar flare event has the possibility of causing severe damage and even death to the space traveler. Nominal spacecraft thicknesses provide little shielding against solar-particle event (SPE) protons. Well-shielded radiation shelters must be provided on exploration spacecraft and on the lunar and martian surfaces. Warning systems need to be developed that can give adequate notice of a forthcoming solar event. There is also concern that weightlessness or reduced gravity will exacerbate the negative effects of radiation. Weightlessness reduces the effectiveness of the immune system that is known to play an important role in lessening the impact of radiation exposure. If an interaction effect should be demonstrated between microgravity and radiation, levels of acceptable dosimetry may need to be adjusted downward, greater protection provided, or some other means devised to lower the risk of space radiation exposure. Specific shielding technologies to be evaluated and developed are waste water uses, propellants, new lightweight composite materials, active electromagnetic radiation shielding, and use of planetary surface materials for outpost dwellings. Selected pharmaceutical radioprotectants also need to be evaluated.

The lunar observatory, Phobos expedition, Mars expedition, and lunar phase of the evolutionary mission pose like threats, while the Mars phase, being much longer, poses a significantly greater threat. All four exploration case studies require that total mission and career radiation dose be minimized and that solar flare warnings be provided.

Life Support. A fundamental life support objective is to develop regenerable systems in order to decrease, and eventually nearly eliminate, the resupply problem. A related near-term need is an accurate monitoring system to assess the quality of recycled air, water, and eventually food, and to detect trends or changes in environmental quality. The Life Sciences Division has in the past developed physicochemical life support systems, and at present is continuing the investigation and development of biological regenerative life support systems. Both physicochemical and biological regenerative systems require continued development to meet the requirements of exploration. Once these systems have progressed to the testing phase, Space Station Freedom will be used to assess their reliability in microgravity, both at a subsystem and at an integrated-system level. Physicochemical systems developed in the past are being integrated and tested by the Space Station Freedom program. Of particular urgency is the requirement to reprocess and reclaim wastewater. Water is massive and essential and expensive to transport. The technology needed to increase the regenerative yield is within reach. A high percentage of water reclamation is a first-order objective being pursued through ground-based research and it is anticipated that most water-reclamation questions will be addressed through planned Space Station Freedom research and testing.

Successful completion of the planned Space Station Freedom life support system and related flight projects should resolve the baseline life support issues for the lunar observatories, the Phobos expedition and the Mars expedition, provided that these missions allow for the inclusion of a large mass of consumables.

The Mars outpost phase of the evolutionary case study presents more direct life support challenges than the other missions. It would be impractical to launch the number of cargo flights needed for full resupply, with redundancy, of all consumables necessary to support a crew of eight for several years. This case study argues for a requirement of a high level of environmental closure for water, air, and food. Such a requirement would call for a biologically regenerative system for the evolutionary case. It also suggests the need to utilize the resources of the lunar and Mars environments as part of those systems.

Space Human Factors. Space human factors refers to the human design requirements and countermeasures necessary to ensure safe, productive, and enhanced crew performance in various space missions. The life sciences program provides a multifaceted approach to understanding basic human capabilities under varying space environments, including perceptual, cognitive, and psychophysiological; and the relationships involved in human-machine interaction, including the important issues of automation and information management. Crew interaction factors, including optimal organizational structures and crew support systems, are also included. Another dimension of the program involves human-operational and human-environmental interactions including mission analysis, crew training and selection, and habitability requirements. This research is supported by human factors modeling, data base, and the use of analog environments. The HiS element of Pathfinder will involve expansion of this program, focusing on extended-duration mission factors. Verification will be achieved on Space Station Freedom through inflight observation and experimentation.

Space human factors is critical to exploration missions. The isolated, remote, and long-duration elements of these missions render the human in the system as important as any hardware element. Automation, human-machine task allocation, and human-telecommunication and telerobotics are issues that permeate all the exploration case studies, although to varying degrees. For the lunar outpost phase of the evolutionary case study, crews remain close to and affected by Space Station Freedom and Earth. In contrast, crews of a Phobos expedition or a Mars expedition or outpost, while also dealing with long-duration effects would in addition be operating with much less help from the ground. As distance from the Earth increases, communication times also increase. The resulting transmission delays have the necessary effect of placing greater responsibility on the crew, particularly in making time critical decisions. The greater autonomy of crews will influence how crews are constituted and trained, as well as what information and other systems are needed to support them. The Mars outpost places the highest demand on space human factors, since it combines remoteness with very long missions of 2 to 3 years.

With the possible exception of the sporadically tended lunar observatories, all other exploration missions require mission task and workload analyses and enhanced crew factors including optimal organizational structures, specialized training and selection criteria, and social and psychological support systems.

3.3.2.2 Enabling Life Sciences Program Timelines

Figure 3.3.2-1 shows the programs and associated timelines in the Life Sciences Division's plan that will be used to develop the requirements for the five essential areas.

The particular areas that are associated with each program are indicated in the parentheses following the program name. The timelines are those projected for the entire program; dates relevant to OEXP requirements may occur much earlier, as will be discussed in section 3.3.3.1.

All essential life sciences areas are supported by the baseline life sciences program and by some aspect of the flight program. Follow-on activity for all areas except advanced medical care is planned under the HiS element of Pathfinder. The Space Biology Initiative will be used to determine requirements for artificial gravity, radiation protection, life support, and space human factors. The extended duration orbiter will provide information needed for development of advanced medical care and zero-g countermeasures, while the EDCO will address both these areas and space human factors. The 1.8–m centrifuge will supply additional information on

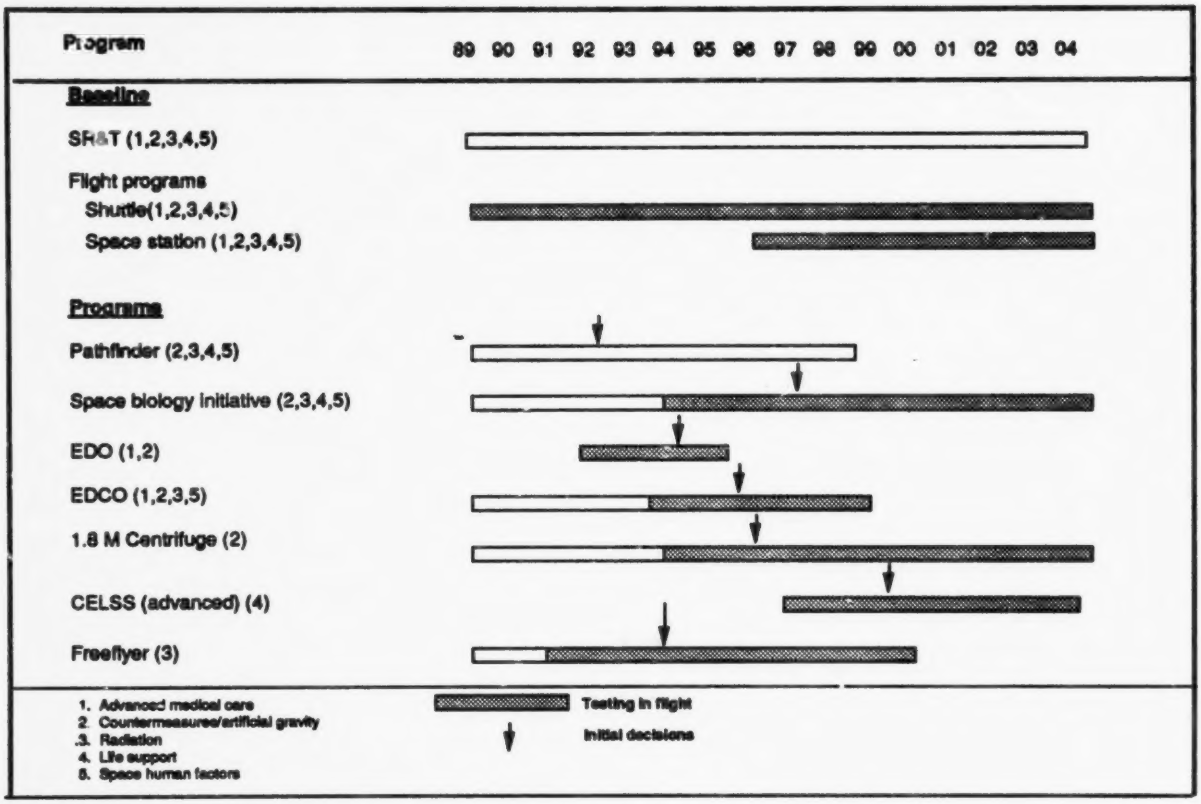


Figure 3.3.2-1.- Life Sciences proposed programs for preliminary planning purposes

artificial gravity effects. The polar freeflyer will provide the dosimetry measurements that will assist in planning radiation shielding requirements.

3.3.2.3 Assumptions of Near-Term Programs and Plans

The near-term programs and plans described in the previous section are predicated both on obtaining adequate access to space and on adequate budgetary support. The near-term plans assume regular life sciences opportunities on Space Shuttle and an opportunity to maintain regular freeflyer polar missions. It is assumed that flight testing of the HMF implementation of the Space Biology Initiative package will be accomplished early on space station. The artificial gravity experimentation will be initiated through the 1.8-m centrifuge to be flown early in the 1990's on spacelab missions. This activity along with an Agency-supported EDCO will be part of phase I Space Station Freedom. Also, it is proposed that life sciences research opportunities, including final verification of concepts for exploration missions, will be implemented in phase II Space Station Freedom.

From a budgetary perspective, the near-term plans include research and development supported by the anticipated Life Sciences Division's budget, augmented to include the OSSA Strategic Plan and also life sciences support from the HiS element of Project Pathfinder.

3.3.3 Prerequisite Program Accommodation of Case Studies

The previous section described both ongoing activities of the Life Sciences Division and the activities outlined in the OSSA Life Sciences Strategic Plan as they apply to establishing requirements for exploration missions. The present section will compare directly the plans and schedules of the Life Sciences Division with the case study requirements outlined by OEXP.

3.3.3.1 Implementation of Four Candidate Case Studies

The schedules for ongoing and planned life sciences activities as compared with OEXP-projected dates are given in figures 3.3.3-1 through -4. Figure 3.3.3-1 provides information on each of the critical areas for the Phobos case study; the other three show similar information for the Mars expedition, lunar observatory, and evolution case studies.

Again, it should be noted that these estimates are for delivery of life sciences requirements and/or prototypes

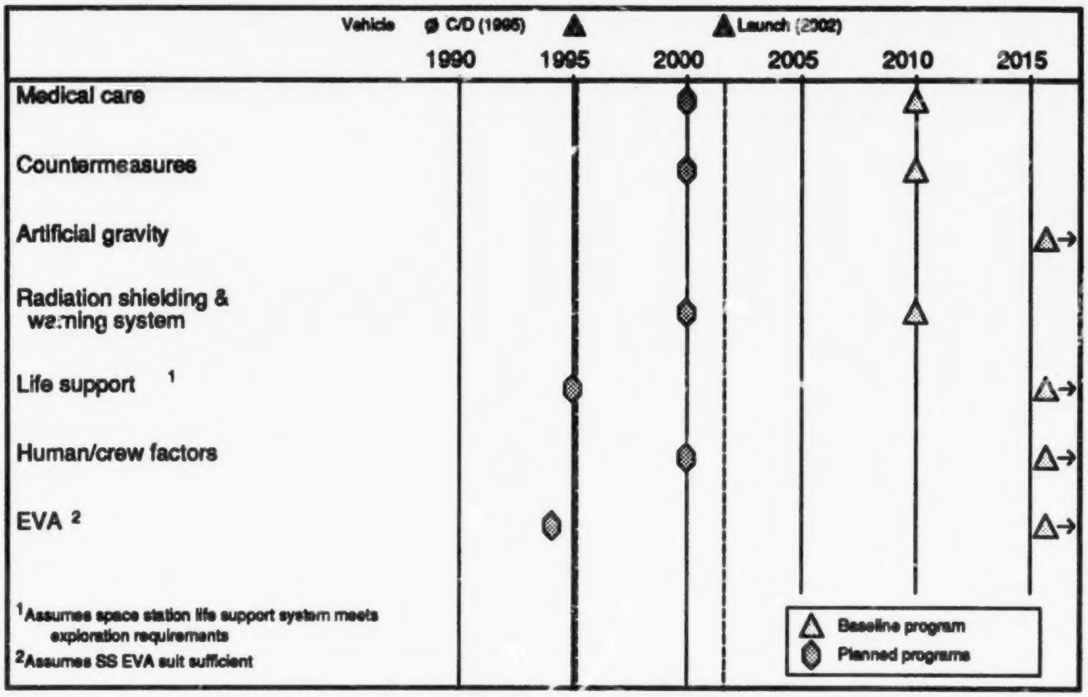


Figure 3.3.3-1.-Phobos systems requirements

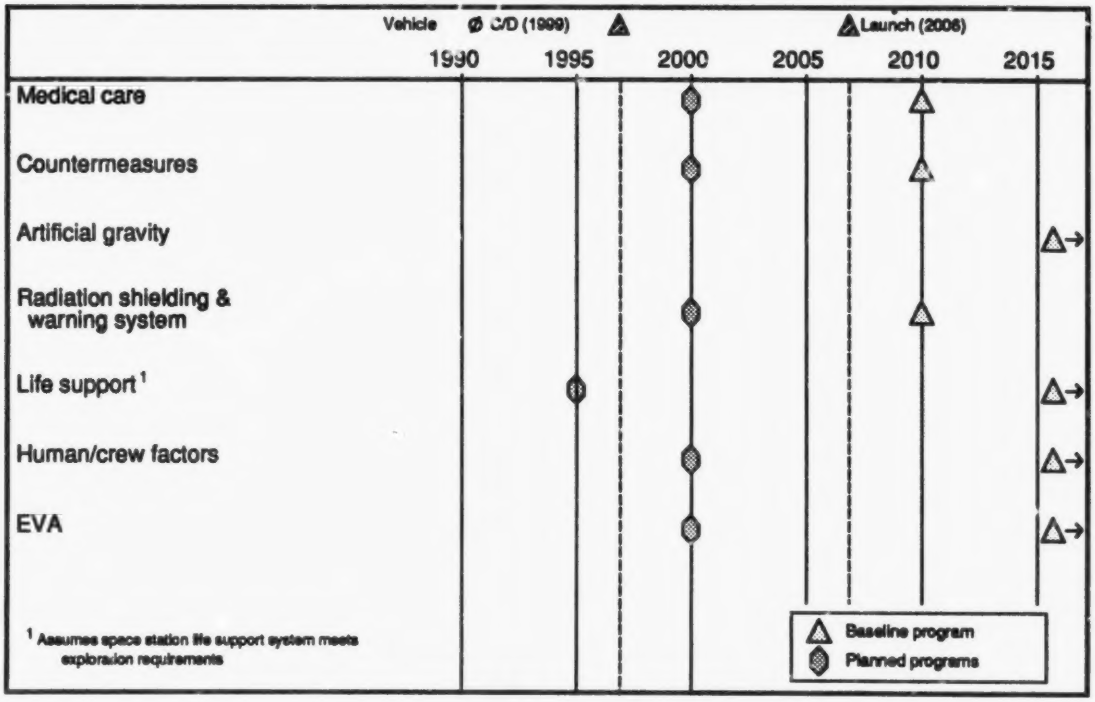


Figure 3.3.3-2.-Mars systems requirements

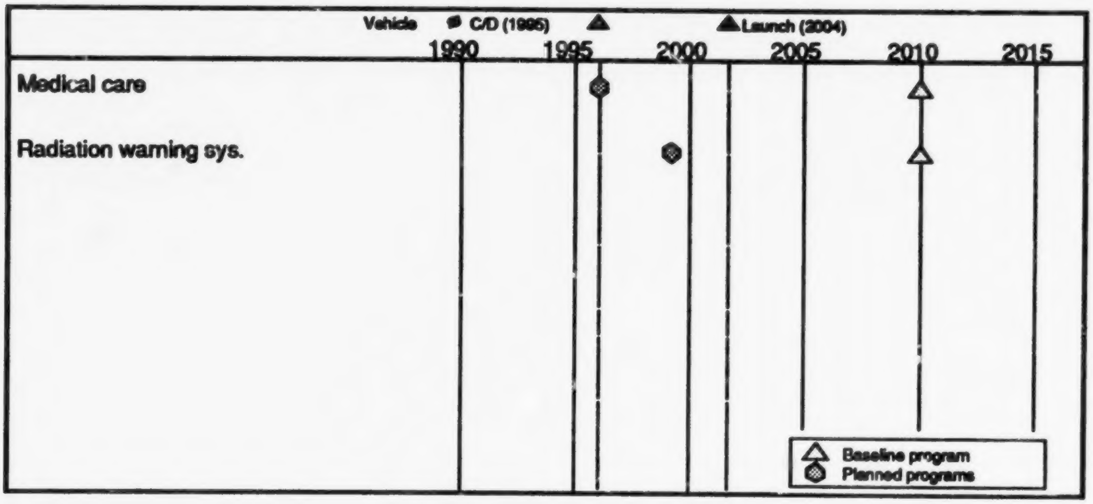


Figure 3.3.3-3.-Lunar observatory systems requirements.

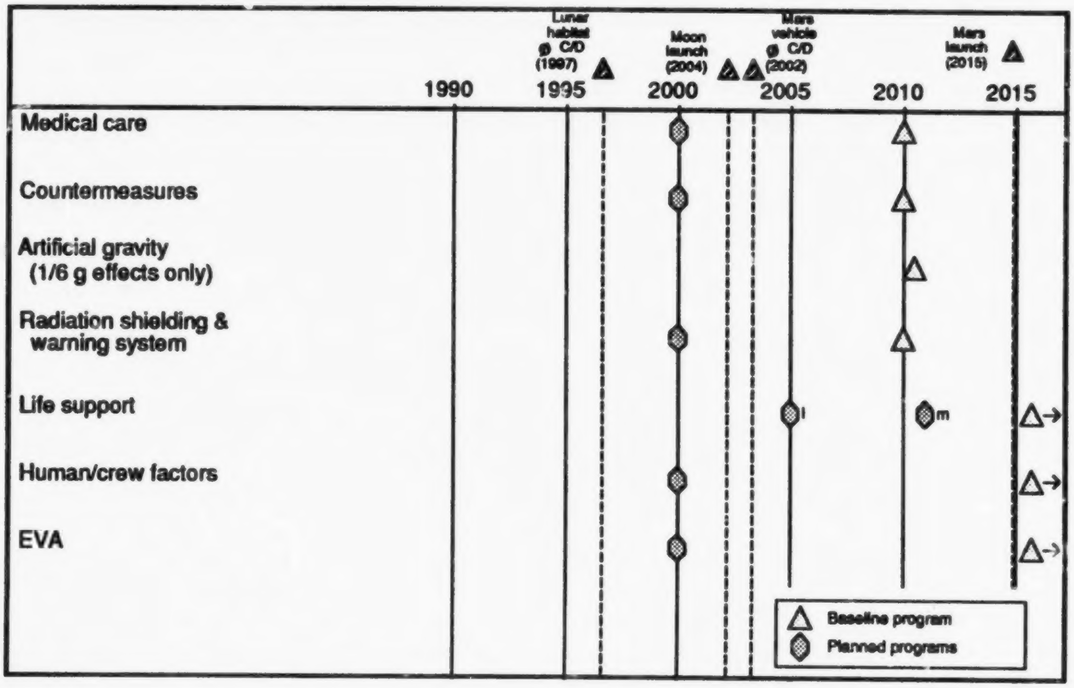


Figure 3.3.3-4.- Moon-Mars evolutionary systems requirements.

of generic and focused technologies, including research hardware and facilities, but do not include the development and implementation of mission-specific flight hardware.

The schedules show that support of the OSSA Life Sciences Strategic Plan is absolutely necessary to accommodate case study needs. The baseline program is inadequate in all cases. It should also be noted that resolution is outstanding for the artificial gravity and advanced life support for the Mars portion of the Moon-Mars evolutionary case. Augmentation to the OSSA Life Sciences Strategic Plan will be required to capture these particular case study needs. Further study is needed of the interactions of the countermeasures program, the artificial gravity program, and the case study vehicle development, as well as of Mars life support requirements, so that compatible programs may be planned. Figure 3.3.3-5 illustrates

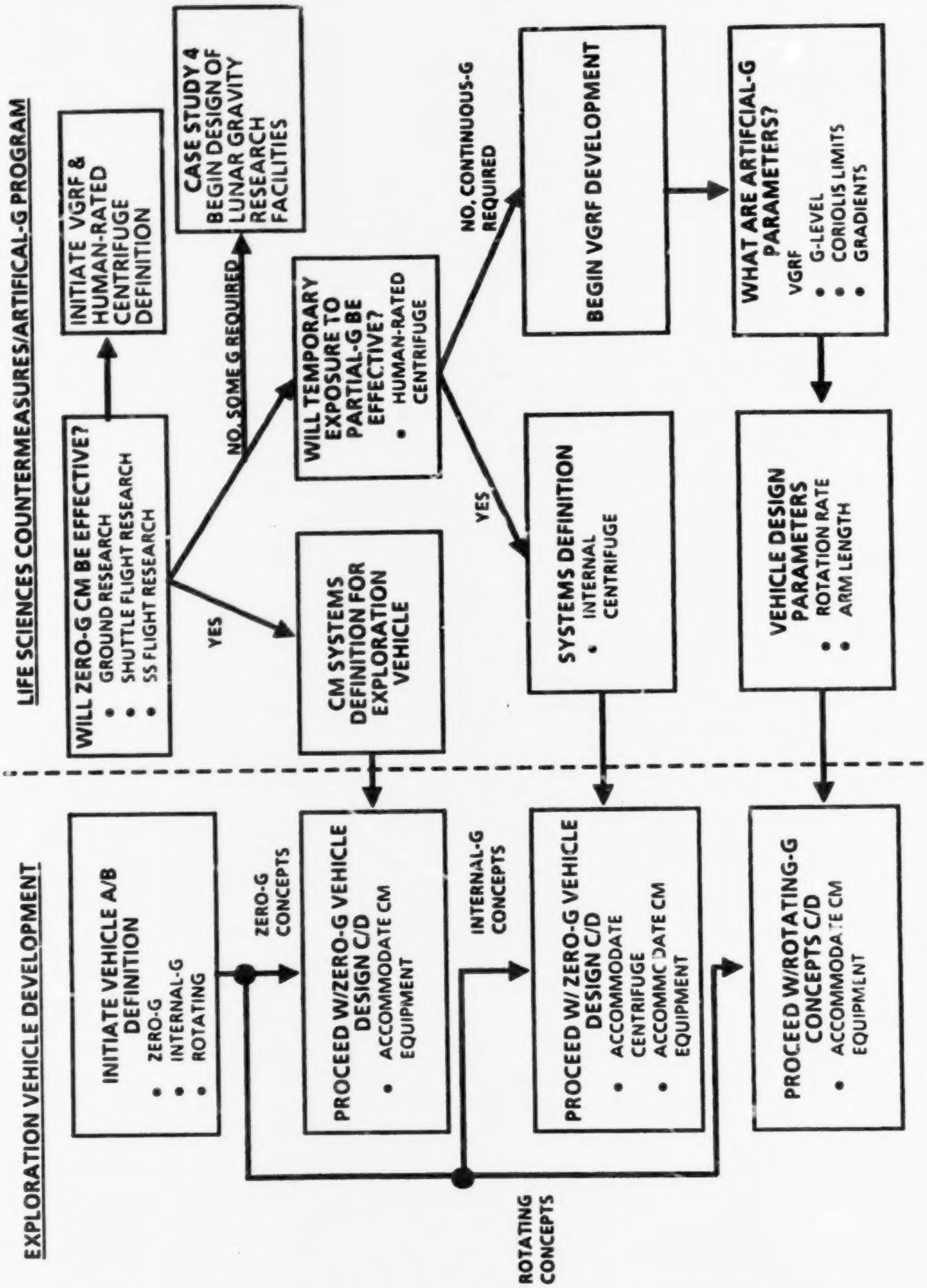

LIFE SCIENCES COUNTERMEASURES/ARTIFICAL-G PROGRAM
WILL ZERO-G CM BE EFFECTIVE?
• GROUND RESEARCH
• SHUTTLE FLIGHT RESEARCH
• SS FLIGHT RESEARCH
INITIATE VGRF & HUMAN-RATED CENTRIFUGE DEFINITION
CASE STUDY 4 BEGIN DESIGN OF LUNAR GRAVITY RESEARCH FACILITIES
YES
NO, SOME G REQUIRED
CM SYSTEMS DEFINITION FOR EXPLORATION VEHICLE
WILL TEMPORARY EXPOSURE TO PARTIAL-G BE EFFECTIVE?
• HUMAN-RATED CENTRIFUGE
YES
NO, CONTINUOUS-G REQUIRED
SYSTEMS DEFINITION
• INTERNAL CENTRIFUGE
BEGIN VGRF DEVELOPMENT
VEHICLE DESIGN PARAMETERS
• ROTATION RATE
• ARM LENGTH
WHAT ARE ARTIFCIAL-G PARAMETERS?
VGRF
• G-LEVEL
• CORIOLIS LIMITS
• GRADIENTS
EXPLORATION VEHICLE DEVELOPMENT
INITIATE VEHICLE A/B DEFINITION
• ZERO-G
• INTERNAL-G
• ROTATING
ZERO-G CONCEPTS
PROCEED W/ZERO-G VEHICLE DESIGN C/D
• ACCOMMODATE CM EQUIPMENT
INTERNAL-G CONCEPTS
PROCEED W/ ZERO-G VEHICLE DESIGN C/D
• ACCOMMODATE CENTRIFUGE
• ACCOMMODATE CM EQUIPMENT
PROCEED W/ROTATING-G CONCEPTS C/D
• ACCOMMODATE CM EQUIPMENT
ROTATING CONCEPTS

Figure 3.3.3-5.- Typical composite strategy for life sciences research & exploration vehicle development.

the type of interactive planning that must be developed for linking the countermeasures and artificial gravity programs with the case studies.

3.3.3.2 Uncertainties

Uncertainties associated with the plans for Life Sciences support of the candidate case studies fall into three categories: budget, access to space, and research and technology development outcomes and further tradeoffs.

Schedules and other plans outlined in this chapter are based on assumptions of a favorable baseline budget augmented to reflect the OSSA strategic plan and a significant HiS/Pathfinder allocation. Yet neither of these requirements is ensured. The recently released NASA Life Sciences Strategic Planning Study Committee findings (the Robbins Report) states: "Life Sciences as it has been is inadequate to support the goals of NASA," a view essentially echoed by the 1988 report of the National Academy of Sciences Space Science Board. However, the 1989 budget activities now being worked through Congress do not support the President's request in this area. The HiS/Pathfinder program also has suffered major losses. Given the budgetary uncertainties associated even with near-term activities, it is difficult to plan for long-term requirements. The schedule information presented in figure 3.3.2-1 will be highly susceptible to revision if budgets are not forthcoming on the schedule anticipated.

Access to space flight is an additional requirement that must be met on a regular and predictable basis in order to support the definition of life sciences requirements for exploration missions. The accommodations on Shuttle, extended duration orbiter, Space Station Freedom, commercially-developed facilities, and launch capability for the freeflyer are all built into the projected data. Like budgetary slips, flight time delays would have significant impact on the projected schedules.

A third area of uncertainty arises from the research itself. Although some requirements (such as 1-year life support capability) can be reasonably well projected, others are more open-ended. An estimate of the requirements that can be delivered assumes an orderly conduct of research (a reasonable assumption) followed by an orderly series of results (a less reasonable assumption). Breakthroughs can occur that have cost-saving implications for design specifications. For instance, observations on Skylab suggest that after the first few days of flight, astronauts may be able to tolerate higher rates of rotation in space than on Earth. Alternately, it is possible that a systematic assessment of artificial gravity could uncover negative side effects. Such an outcome would necessitate the rethinking of how deconditioning effects could be managed on very-long-duration flights. In addition to the effects of radiation and artificial gravity, environment uncertainties that may need further investigation include dust, chemicals, and possible mutation of terrestrial organisms.

3.3.3.3 Case Study Opportunities for Basic Science

In addition to the enabling technology programs, the Life Sciences Division conducts a number of programs in the basic sciences that have direct relevance to exploration missions. The life sciences basic research programs are fundamental to understanding the role that gravity plays in plant and animal life on Earth and elsewhere; to exploring the origin, evolution, and distribution of life in the universe; and to understanding the relationship between biological systems and an evolutionary biosphere. Current programs in gravitational biology include cell biology, gravitational perception and sensing, development of organisms, and adaption to gravity. Exobiology research is directed towards understanding the evolutionary pathways leading from the cosmic evolution of biogenic compounds to the evolution of advanced life, and towards finding evidence of this evolution elsewhere in the universe. The understanding of the biosphere is based on understanding the interaction of biological and physical/chemical processes on a global scale and provides a basis for monitoring changes in the Earth's habitability. Each of these programs could benefit from advanced missions of exploration.

A lunar surface outpost could provide a stable reduced-gravity laboratory for a range of investigations from microbiology to whole-plant studies. Mars could serve as a similar outpost in studies of higher level gravitational force.

Exobiology interests would be addressed by exploration of the solar system, as represented by a mission to Phobos or Mars. Even in the absence of evidence of existing life, a more thorough examination of the Phobos or martian surface would provide evidence concerning molecular evolution leading to life in this solar system. A link in the argument for extraterrestrial intelligence is the prevalence of planets around other stars.

A lunar outpost would provide a laboratory from which to conduct a planetary search.

An observatory on the far side of the Moon is also considered by some to be the ideal location for a continuing radio telescope search for extraterrestrial signals from other intelligent civilizations. The lunar surface would be an ideal place from which to collect interplanetary dust particles and to study the processes that led to the formation of the solar system as we know it.

It is widely believed that the surface of Mars once held large quantities of water, and that the early climate of Mars resembled that of the early Earth. While this climate prevailed on Mars, life was burgeoning on Earth. The possibility that life arose on Mars in the same time frame as it did on Earth makes Mars a focus of the search for life, extant or extinct, off the Earth. An investigation of the changing climate of Mars, and its possible interrelationship with a martian biota, is of interest to both the exobiology and the biospherics research programs. Determining how, or if, the martian system cycles materials and gases would provide a comparative link in understanding the biospheric possibilities of our own planet. The evolutionary case study leading to an extended stay on the martian surface would serve this research need.

These basic research experiments, though important in themselves, are not totally divorced from the enabling science and research that are the focus of this chapter. For instance, a general understanding of micro- or reduced-gravity effects supports directed research projects such as animal and human tolerances to microgravity and the potential for food growth in microgravity. Additionally, the information needed to assess the exobiological potential of Mars, the Moon, or Phobos is the same information needed to begin the use of in situ resources to provide for an extended human presence in space.

3.3.3.4 Critique of Case Studies

Life sciences requirements differ from those of, for instance, propulsion or transportation systems, in that living systems, although flexible in many ways, are totally unaccommodating in others. For instance, if oxygen is lacking in the environment, oxygen must be supplied — there are no alternate solutions or workarounds. For this reason, a life science critique of case studies is less a statement of preferred methods of solutions than a statement of how various case studies mesh with planned life science developments.

The Lunar Observatory Case Study presents few new challenges to the life sciences. The Apollo missions established the need to deliver humans to the lunar surface in good condition and to provide the wherewithal to support them for periods of from 1 to 3 days. This requirement included the need for EVA's of over 7 hours at a time, with one crew spending over 22 hours exploring the lunar surface during their three EVA's. The lunar observatory case study requirement for 14-day stays with twelve 6-hour EVA's per mission is viewed as a incremental increase over the requirements of Apollo and, theoretically at least, should present no major obstacles. For this case study, what is necessary is to develop radiation protection and warning systems for use on the lunar surface. However, it would be desirable to develop more appropriate EVA systems; e.g., suits that allow greater use of the legs and therefore permit greater mobility and easier hand movement than is available with the Apollo suit. Significant effort is also required to develop heads-up displays and to foster ease of use and long-term maintenance. In terms of scheduling, there is no conflict with the life sciences requirements.

The Human Expeditions to Phobos and Mars missions are alike in the challenges they present to the life sciences. Both provide for crews aloft for periods of 500 days, longer than we are presently capable of supporting in terms of health care, life support, crew operations, or logistics. Although most of these issues should be amenable to investigation, all of them need to be worked and resolved. One question that does distinguish the Phobos and Mars expedition missions is the size of the crew. How optimal crew size (most productive grouping) relates to minimal crew size (large enough only to accomplish the essential tasks) needs to be explored. It would be highly desirable to gain further understanding of long-duration effects on crews of eight members, the proposed crew size of the transportation phase of the human exploration of Mars. Workload analysis, crew organizational systems, and crew training are all interactive factors in understanding this issue. In both the Phobos and the Mars case studies, a highly significant question is whether crews can withstand exposure to weightlessness for more than a year without some form of artificial gravity to offset deconditioning effects. One possibility is that, without artificial gravity, those crew members who land on Mars will have difficulty operating on the 1/3-g martian surface. However, as a result of their 20-day stay on Mars, these astronauts could return to Earth in better condition than either the Phobos astronauts or the Mars missions astronauts who remain in Mars orbit. The issue of deconditioning effects, countermeasures, and the potential need for artificial gravity is a serious one and, as outlined above, will require new activity in order to be adequately addressed in time to meet the current OEXP schedules.

The Lunar Outpost to Early Mars Evolution Case Study is the most complicated of the four options, calling for a continuing human presence on the lunar surface and three manned missions to Mars of more than 3 years' duration. Ostensibly this case study presents the greatest demands on the life sciences. However, it also contains within itself some means of addressing, at one stage in the mission, issues that must be resolved for a later stage in the mission. Also, the proximity of crews during the lunar phase offers some real advantages. For instance, rescue is possible, and changeout of crew members, if required, could be accommodated. The Mars phase of the mission represents a dramatic increase in the need for life sciences support. We have only primitive understanding of how to prepare crews and systems for 2 to 3 years of autonomous operations. This phase of the mission also dramatically increases crew exposure to micro- and reduced gravity. However, in this case study, all crew members experience the potentially beneficial effects of the 1/3-g martian surface. Since this case study provides a 1/6-g lunar laboratory opportunity on which to assess the long-term effects of a reduced-gravity environment, we will know in advance how well the 1/3 g of Mars might help offset deconditioning effects. Because of the long duration of exposure, this case study also presents a high radiation threat. However, the most important issues involve life support. Although possible, it would seem unreasonable to embark on the Mars phase of this mission without significant advances in physical-chemical and biological regenerative life support processes. Flights of this duration demand a significant level of environmental closure. According to present plans, the Life Sciences Division schedule would fall short in several areas of the projected date suggested by OEXP. However, no new Life Sciences Division work is proposed in this area at this time because the availability of the lunar outpost phase would provide the testbed on which needed developments could be pursued and would allow determinations to be made close to the OEXP dates.

3.3.4 Support Required from Other NASA Organizations

Near-term projects and plans constitute precursor requirements to exploration missions and require the continuing and timely support of other NASA Headquarters organizations, specifically, codes M, EL, ES, R, and S. Since the baseline precursor studies are generally similar, organizational support requirements for the several case studies will be considered together.

Code M will be asked to supply freeflyer launch capability to polar or geosynchronous orbit in support of radiation dosimetry measurements. Also required will be dedicated middeck accommodations on Shuttle and experimental access to the extended duration orbiter.

The support of code EL will be needed to fly radiation detectors on planetary missions. This equipment is necessary to establish the experiential data from which tolerance levels, countermeasure requirements, and warning systems can be developed.

The data developed as a result of the freeflyer flights must be converted into a reliable and accurate warning system. The efforts of code ES will be needed to construct a warning system that will allow the astronauts on the lunar, Phobos, or martian surface to reach shelter prior to an anticipated solar radiation event.

The HiS element of Pathfinder is a joint effort of code E and code R. Essential to the plans for all case studies presented here is support from code R and Project Pathfinder for all critical areas. If the Lunar Outpost to Early Mars Evolution Case Study is the mission of choice, the cooperation of code R will also be required to help plan for the mining and utilization of the martian environment to meet life support requirements.

Code S will play a very important role in accommodating the life sciences plans that form the bases for exploration missions. For all case studies, flight support from the Space Station Freedom HMF, the EDCO, and the Space Biology Initiative package will be required. For the Phobos, Mars, and Mars outpost missions, support of artificial gravity experiments in the form of flying the 1.8–m centrifuge and, potentially, a variable gravity research facility will be required. It is likely that life sciences requirements will reach a level requiring a dedicated life sciences module on Space Station Freedom as early as 1998.

3.3.5 Conclusions

The activities planned by the Life Sciences Division over the next several years satisfy the scope and generally meet or approximate the schedules outlined in OEXP's candidate case studies. However, in all areas, meeting or approaching the proposed schedules will require a very aggressive program, representing a quantum leap in research and development activity over the present baseline program. Such an aggressive program is already contained in the combined OSSA Strategic Plan and the HiS element of Project Pathfinder. This program would have to be implemented. Two exceptions would still remain to the confluence of the Life Sciences Division and OEXP schedules: the development of requirements for artificial gravity (for which new work is proposed) and the development of life support capabilities for the Mars outpost phase of the evolutionary case study (for which investigations would be conducted during the lunar outpost phase.) In addition to the research and engineering requirements noted, the support of other Headquarters organizations, particularly in providing early access to space flight testing, is essential.

In addition to requirements necessary for exploration missions, the Life Sciences Division has a deep interest in the basic science research that could be accomplished through exploration missions. Whether in furthering the understanding of gravity effects exploring the origin and evolution of life in the universe, or charting variation on global ecology, the Life Sciences Division stands ready to take full advantage of basic science opportunities made possible through exploration missions.

3.4 EXPLORATION IMPACTS TO AERONAUTICS AND SPACE TECHNOLOGY

3.4.1 Role of the Office of Aeronautics and Space Technology (OAST) in Manned Exploration

In FY 1989, the two-part structure of the space research and technology (R&T) program, the R&T base and the focused programs, is continued. The R&T base contains generic, fundamental research; the focused programs include the Civil Space Technology Initiative (CSTI) and the Pathfinder initiatives. CSTI is intended to remedy gaps in the U.S. space technology program and help restore the Agency's technical strength, contributing to a world leadership role. The CSTI program will provide advanced technologies to support three specific areas of near-term, high-priority missions, near-term transportation needs, enhanced operations in low earth orbit (LEO), and science operations. CSTI supports research and technology in propulsion, vehicles, information, large structure and control, power, and automation and robotics, concentrating primarily on application for LEO operations. Project Pathfinder is a NASA technology initiative to develop technical understanding and hardware in a set of critical areas which will make possible future national decisions regarding exploration of the solar system consistent with the President's space policy and the primary directives of OEXP. Pathfinder does not, in itself, represent a commitment to a particular mission, but rather a commitment to provide a sound engineering basis for a future pathway decision. To a lesser degree other OAST (code R) programs also support solar system exploration.

Project Pathfinder is a focused research and technology program that will enable a broad set of new missions in space exploration and strengthen the technology base in support of the civilian space program. Pathfinder extends the foundation established under CSTI. By bridging critical technology gaps, CSTI enhances access to Earth orbit and supports effective operations and science missions in orbit. Pathfinder, with a longer term horizon, foresees a future that builds on the Space Shuttle and Space Station Freedom and addresses technologies that will support a range of future space missions, including a return to the Moon to build an outpost, piloted missions to Mars, and continuing exploration of Earth and other planets. The program's objective is to develop, within a reasonable timeframe, those emerging and innovative technologies that will make possible both new and enhanced missions and system concepts. Early proof-of-concept testing for mission-critical engineering designs will be an important element of the overall Pathfinder program. These demonstrations will directly support continuing evolution and maturation of mission plans. The elements of the Pathfinder program are currently partitioned into four thrusts as shown in table 3.4.1-I. The program addresses opportunities afforded by the advancement of high-leverage technologies in the major program thrusts.

3.4.1.1 Exploration Technology

The technologies included in exploration are related to acquiring scientific knowledge and technical understanding at mission sites on the Moon and Mars. Specific objectives include development of the capabilities needed to precede piloted flights to Mars and for the construction of a lunar outpost. Program elements are planetary rover development; remote sample acquisition, analysis, and preservation technology development; surface power research; and optical communications research.

3.4.1.2 Space Operations Technology

Space operations technology deals primarily with the lunar outpost, with piloted missions to Mars, and with operations in Earth orbit. For lunar and Mars missions, this program will address critical technologies for preparing to depart Earth orbit, for performing mission tasks on arrival at surface sites, and for safe return from the Moon or Mars. For Earth orbit operations, this program will greatly extend the capability to maintain an infrastructure and to support major new science missions. Specific objectives include extensive capabilities for in situ materials processing, fabrication, and assembly and repair of massive and complex systems in Earth orbit and at lunar and martian orbits and surfaces. Program elements include autonomous rendezvous and docking technology; resource processing pilot plant research, in-space assembly and construction research; cryogenic fluid depot technology and space nuclear power technology (SP-100).

TABLE 3.4.1-I.- PROJECT PATHFINDER

Exploration Technology	Operations Technology
Planetary rover Sample acquisition, analysis, and preservation Surface power Optical communications	Autonomous rendezvous and docking Resource processing pilot plant In-space assembly and construction Cryogenic fluid depot Space nuclear power (SP-100)
Humans-in-Space Technology	**Transfer Vehicle Technology**
Extravehicular activity suit Human performance Closed-loop life support	Chemical transfer propulsion Cargo vehicle propulsion High-energy aerobraking Autonomous lander Fault-tolerant systems

3.4.1.3 Humans-in-Space (HiS) Technology

Existing technologies cannot be scaled to meet human requirements over long missions requiring self-sufficiency. HiS will address the technology solutions, based on identified human requirements, that make it feasible and productive to send astronauts on lengthy missions to the Moon and Mars. The objectives of the HiS thrust of Pathfinder are to (1) determine the critical engineering technology requirements and develop and validate the technology options to enable human self-sufficiency in space, an OAST task, and (2) conduct research to determine the human requirements which must be met by engineering technology solutions and develop biological support systems for self-sufficiency, an Office of Space Science and Applications (OSSA) task. Program elements are (1) extravehicular activity (EVA) suit; (2) human performance, consisting of human factors, psychosocial behavior, and human health technology; and (3) closed-loop life support systems, both physical/chemical and biologically based.

3.4.1.4 Transfer Vehicles Technology

Transfer vehicles technology will support transportation to and from geostationary Earth orbit (GEO), the Moon, Mars, and other planets. Specific objectives include significant reduction in the mass that missions require to be launched into LEO and in transit, as well as reductions in the time required for transit. The key technology program elements are chemical transfer propulsion research, cargo vehicle propulsion development, high-energy aerobraking development, autonomous lander development, and fault-tolerant systems development.

The goals of Project Pathfinder are to develop critical technology opportunities for a range of future solar system exploration missions, to support a national decision regarding future missions in the early 1990's, and to support broad U.S. civil space technology leadership. To achieve these goals, the Pathfinder management system and the content of the individual element programs have been designed to address five specific objectives.

a. Produce initial critical research results and validate key capabilities by the early 1990's.

b. Achieve necessary levels of readiness and transition technologies to mission users beginning in the mid-1990's.

c. Define and achieve the right balance between more basic research and focused demonstrations.

d. Coordinate Pathfinder research and technology with other NASA offices and support ongoing NASA mission studies.

e. **Build a partnership among NASA, U.S. industry, and universities in the implementation of Pathfinder programs.**

The complete description of the Pathfinder program is contained in the Project Pathfinder Program Plan, which defines the overall project as well as the element programmatic objectives, the schedules and milestones, the deliverables, and the management structure for each element. The Pathfinder program, as submitted to the Congress, was initiated at $100M in FY 1989 and had a 5-year runout of $840M. Although the expected FY 1989 budget levels are expected to be somewhat below the $100M level, current planning assumes that the out-year budgets will stay at the proposed levels.

3.4.2 Case Study Needs Assessment

3.4.2.1 Critique of Case Studies

Following a preliminary overview of four case studies under examination by OEXP, the technology requirements for each case have been identified, and the applicability of the OAST program to the requirements has been assessed. Original OEXP prerequisite needs have been updated based on detailed discussions between OAST and OEXP as shown in table 3.4.2-I. The table has been annotated to indicate where the requirements, from a technology perspective, depart from those based on a mission perspective. Asterisks have been inserted where OAST believes a requirement needs to be added. Shaded ellipses indicate the requirements that are covered by the OAST technology program. The details of the programmatic coverage are discussed in section 3.4.3. Although almost all the requirements are covered and most of the required technologies would be developed approximately when needed, in some cases the scope and pace of the programs does not match the detailed mission schedules. It is expected that the detailed comparison of the technology program schedules and the OEXP case studies milestones will be analyzed in great detail over the next few years. The analysis will include comparing projected states of technology readiness achievable within planned budgets with the levels of technology readiness required to support the phase C/D initiations in the human exploration case studies.

Tables 3.4.2-II through -V show Project Pathfinder technology readiness levels and their required dates, along with dates currently planned by OAST. Disconnects in programs, technology readiness levels, and planned dates are indicated by an asterisk (*).

An initial comparison of mission milestones and technology readiness has identified three technology areas — life support, aerocapture/entry, and cryogenic fluid management — that are inconsistent with three of the

TABLE 3.4.2-I.- TECHNICAL REQUIREMENTS MATRIX

	Scenario			
Technology Requirement	Case 1	Case 2	Case 3	Case 4
Propulsion				
Launch	HLLV?			
Orbit Transfer:				
Mars	⍟	Ⓧ		Ⓧ
Lunar			Ⓧ	Ⓧ
Electric		Ⓣ		Ⓣ
Ascent		X	X	X
Descent		X	X	X
Power				
Space Transfer Vehicle		Ⓧ		Ⓧ
Electric Prop		Ⓣ		Ⓧ
Rovers	Ⓧ	Ⓧ	Ⓧ	Ⓧ
Unmanned				
Piloted		Ⓧ	Ⓧ	X
Surface facilities				
Temp Surf Lander		Ⓧ	Ⓧ	Ⓧ
Outpost			Ⓧ	Ⓧ
Base				Ⓧ
Life Support				
Physical chemical	⍟	⍟	⍟	Ⓧ
Biological				Ⓧ
Human performance				
Man machine Interactions	⍟	Ⓧ	Ⓧ	Ⓧ
Extravehicular Activity	⍟	Ⓧ	Ⓧ	Ⓧ
Aerobraking/Aerocapture				
Planet capture	Ⓣ	Ⓧ		Ⓧ
Planet Entry		Ⓧ		Ⓧ
Earth capture	⍟	Ⓧ	Ⓧ	Ⓧ

Legend

X -- OEXP technology development required
T -- OEXP technology being considered in trade study
TBD -- To be determined
○ -- Technology covered by OAST program
★ -- Technology need identified by OAST

TABLE 3.4.2-I.- Concluded

Technology Requirement	Scenario Case 1	Case 2	Case 3	Case 4
Materials	TBD	TBD	TBD	TBD
Structures				
Surf assem. & const			X	X
Onorbit assem. & const	ⓧ	ⓧ	ⓧ	ⓧ
Artificial-g spacecraft		Ⓣ		ⓧ
Radiation shielding	ⓧ	ⓧ	ⓧ	ⓧ
Automation & robotics			ⓧ	ⓧ
Automated rend & docking	ⓧ	ⓧ		ⓧ
Info. processing & cost	TBD	TBD	TBD	TBD
Sensors & instruments	TBD	TBD	TBD	TBD
Rovers				
Unmanned science rover	ⓧ	ⓧ	ⓧ	ⓧ
Manned rover		ⓧ	ⓧ	ⓧ
Unmanned rover at support base				ⓧ
Utility rover				ⓧ
Resource utilization				ⓧ
Surface science	ⓧ	ⓧ	ⓧ	ⓧ
Cryogenic Fluid management	ⓧ	ⓧ	ⓧ	ⓧ
Artificial Gravity Spacecraft		Ⓣ		Ⓣ
Surface Launch Systems				TBD

TABLE 3.4.2-II.- TECHNOLOGY NEEDS AND PLANS FOR SURFACE EXPLORATION

Technology Need	Readiness levels(f) & Date				Levels & Dates Planned	Current Programs
	Case 1	Case 2	Case 3	Case 4		
Short-range robotic rover		level 5 1997	level 5 1995	level 6 1997	level 5 1994	Pathfinder/PR
Long-range robotic rover		level 5 1997		level 6 2001	level 5 1998*	Pathfinder/PR
Unpressurized piloted rover		level 5 1997	level 5 1995	level 6 1997	level 5* 1998*	Pathfinder/PR
Pressurized piloted rover				level 6 2001	level 5* 2001	Pathfinder/PR
Cargo-carrying operations vehicle			level 5 1995	level 6 1997	N/A	No specific pgm.*
Construction operations vehicle		level 4 1997	level 4 1993	level 4 1997	N/A	No specific pgm.*
Mining (& equipment) operations vehicle			level 5 1995	level 6 1997	N/A	No specific pgm.*
Precision landing & hazard avoidance	level 5 1996	level 5 1996	level 5 1996	level 5 1996	level 5* 1998*	Pathfinder/AL
Advanced data & image processing	level 5 1996	level 5 1996	level 5 1996	level 5 1996	level 4 1995-1998	Pathfinder/Ph
Phobos exploration system (vehicle)	level 6 1995				level 5* 1998(robotic)*	No specific prg.* (some PR application)
Surface power (non-nuclear)		level 5 1997	level 5 1995	level 5 1997	level 5 1998*	Pathfinder/SP
Mobile surface power		level 5 1997	level 5 1995	level 6 1997	N/A* (related to SP)*	No specific pgm*
Surface science technologies	level 5 1996	level 5 1997	level 5 1997	level 5 1998	level 5 1998*	Pathfinder/SAAP (no observatory tech)*

* OEXP needs and current program plans diverge.

TABLE 3.4.2-III.- TECHNOLOGY NEEDS AND PLANS FOR IN-SPACE OPERATIONS

Technology Need	Readiness levels (f) & Date				Levels & Dates Planned	Current Programs
	Case 1	Case 2	Case 3	Case 4		
Component-level onorbit assembly		level 6 1998	level 6 1995	level 6 1997	level 5* 1998*	Pathfinder/ISAAC
System-level onorbit assembly		level 6 1998	level 6 1995	level 6 1997	N/A	Pathfinder/ISAAC
Vehicle processing onorbit assembly		level 6 1998	level 6 1995	level 6 1997	Post-2010*	CSTI/A&R
Autonomous rendezvous & dock.	level 6 1995	level 6 1997	level 6 1995	level 6 1996	level 5 1996*	Pathfinder/AR&D (no umbilicals focus)*
Storage of cryogenic fluids	level 5 1995	level 5 1997	level 5 1995	level 5 1997	N/A	Pathfinder/CFD (no storage focus)*
Transfer of cryogenic fluids	level 5f 1995	level 5f 1996	level 5f 1995	level 5f 1996	level 2/3* 1993*	Pathfinder/CFD (limited options)
High-rate space communications	level 7 1999	level 7 1999		level 7 1999	level 7 1999	Pathfinder/OC
Space nuclear power (surface use)			level 5 1995	level 5 1997	level 5 2002*	Pathfinder/SP-100
In situ propellant production				level 6 1996	level 4* 1996*	Pathfinder/RPPP (No automation)*
In situ materials fabrication				level 6 1999	level 4 1996	Pathfinder/RPPP (No automation)*
Artificial gravity vehicle systems				level 5 1997	N/A	No specific pgm.*

* OEXP needs and current program plans diverge.

TABLE 3.4.2-IV.- TECHNOLOGY NEEDS AND PLANS FOR HUMANS-IN-SPACE

Technology Need	Readiness levels (f) & Date				Levels & Dates Planned	Current Programs
	Case 1	Case 2	Case 3	Case 4		
In-space EVA suits		level 6 1998	level 6 1995	level 6 1997	N/A	No specific prgm.*
Planet surface EVA suits		level 6 1998	level 6 1995	level 6 1997	level 5* 1998* suit	Pathfinder/EVA (with PLSS)
In-transit life support	level 6 1997	level 6 1997		level 6 2002	level 6 1997	Pathfinder/P-C CLLSS
Initial surface life support	level 6 1997	level 6 1997		level 6 1997	level 6 1997 CLLSS	Pathfinder/P-C (no PLSS)
Advanced surface life support				level 6 2001	level 6 1999	Pathfinder/P-C CLLSS (&CELSS)
Advanced human-machine interfaces	level 6 1996	level 6 1996	level 6 1996	level 6 1996	level 5* 1998*	Pathfinder/HF

* OEXP needs and current program plans diverge.

TABLE 3.4.2-V.- TECHNOLOGY NEEDS AND PLANS FOR SPACE TRANSFER

Technology Need	Readiness levels(f) & Date				Levels & Dates Planned	Current Programs
	Case 1	Case 2	Case 3	Case 4		
Aerobraking (Earth from Moon)			level 5f 1994	level 5f 1995	level 5f	CSTI/AFE
Aerobraking (Earth from Mars)	level 5f 1994	level 5f 1996		level 5f 1995	level 5* 1995	Pathfinder/HEAb (no flight demo)
Aerobraking (Mars from Earth)	level 5f 1997	level 5f 1997		level 5f 2002	level 5* 1994	Pathfinder/HEAb (no flight demo)
STV chemical propusion		level 6 1997	level 6 1995	level 6 1996	N/A	No specific pgm.*
STV/Phobos chemical propulsion	level 6				N/A	STS-derived
Ascent/descent chemical propulsion		level 6 1997	level 6 1995	level 6 1996	level 5* 1994	Pathfinder/CTP (No Mars ascent)
Upper stage chemical propulsion		level 6 1996	level 6 1996	level 6 1996	level 5* 1994	Pathfinder/CTP
STV Nuclear-electric prop.				level 5 1996	level 5/1998 & N/A	Pathfinder/CVP (No cargo reactor)*

* OEXP needs and current program plans diverge.

planned case study mission schedules. Other focused technology developments have been identified as necessary for individual missions: EVA for the human-tended lunar observatories mission and autonomous rover for a manned Mars expedition.

The Phobos case study assumes use of the Space Station Freedom life support system. The mass and risk penalties associated with this technology, for long-duration missions to the vicinity of Mars, are believed to be prohibitive, and need to be studied in considerable depth. The next generation of life support technology will continue to analyze the potential approaches for closing water, air, and waste cycles. It will also perform long-term system tests to ensure that there is not a buildup of toxic materials over extended missions. The reliability of these systems also needs to be demonstrated. The next generation of technology could be established by the year 2000, if the current program is sufficiently augmented. Even if missions were initiated before 2000, interim products of the technology program could potentially be incorporated, and the additional technical flexibility would have substantial benefit to the missions.

All of the human exploration case studies make extensive use of aerocapture and aeroentry. They require the capability for aerocapture at Mars and for high energy aeroentry on return to Earth. The OAST program plan currently includes an aerobraking flight experiment in FY 1993 that would demonstrate the technology necessary for aerocapture on return to Earth from the Moon. Pathfinder lays the engineering foundation for the higher velocity regime necessary for return from Mars, but does not include a flight experiment. The definitive set of experiments could be completed by FY 1998 if sufficiently funded. An alternative is to use the Mars Rover Sample Return (MRSR) mission as an engineering demonstration of both aerocapture at Mars and aeroentry on return to Earth. A mission alternative would be to add to the Mars return capsule a propulsive stage capable of reducing the capsule's velocity to lunar return levels. The implications of this option on the transportation requirements of the Mars missions need to be quantified.

The third long-lead-time technology is cryogenic fluid management. Both the lunar and Mars missions plan to assemble transportations systems in LEO. The Mars missions utilize multiple vehicles, which rendezvous in Mars orbit and transfer propellant required for the return trip. The duration of the Mars missions is on the order of two years. For all of these missions it is very important to make accurate determinations of propellant levels. Prediction capability will greatly affect the propellant margins, which in turn will greatly affect over-all mission mass and launch requirements. All aspects of long-term storage, gauging, and transfer in need of zero-gravity demonstration in LEO could be completed by FY 1996 if the necessary budget could be made available.

Two technologies need augmentation for specific case studies. Human expeditions to Mars case would probably require focused development of an autonomous rover, which could be used to maximize the return from the MRSR precursor. A highly autonomous rover could be demonstrated by FY 1998. The human-tended lunar observatory case would probably require the development of a lunar EVA suit. Sufficiently funded, it could be completed as required by FY 1996.

In several other areas, requirements are so preliminary it is very difficult to determine, in more than a generic way, what technology advances are re-quired: specifically, materials, automation and robotics (A&R), information processing, and sensors for science instruments. A&R, information processing, and sensor technologies tend to be very mission-specific and will also surface later as the individual missions are analyzed in greater detail. A number of requirements clearly are not covered, including ascent and descent propulsion and power for piloted rovers. There is no propulsion program focused on ascent and descent application; however, this is one area where Apollo and Viking experience may be directly applicable. Special, yet-to-be-identified materials problems are sure to surface for various subsystems and applications. Pathfinder does not include surface assembly and construction programs and mining and processing technologies because the case studies lack sufficient maturity in these areas to identify the particular system and subsystem technologies required.

3.4.2.2 Schedules

In figure 3.4.2-1, schedules for major technology demonstrations are shown. These schedules are based on planned Pathfinder funding; changes in funding levels will impact the technology demonstrations.

3.4.3 Prerequisite Program Accommodation of Case Studies

Propulsion. The OAST advanced space propulsion technology program includes a broad-based R&T program effort, the CSTI earth-to-orbit (ETO) and booster technology programs, and two relevant Pathfinder program elements: chemical transfer propulsion and cargo vehicle (electric) propulsion. These activities are directed at providing the technology advancements essential to the projected OEXP missions. The focus of the chemical transfer propulsion (CTP) element of Pathfinder is to validate and extend design and analysis tools previously developed in the R&T base program for advanced liquid oxygen/liquid hydrogen (LOX/LH$_2$) expander cycle engines at the major susbsystem and breadboard engine system levels. In the cargo vehicle propulsion program, the focus is on electric propulsion devices operating at very high power levels (>2MW), including both ion and magnetoplasmadynamic (MPD) thruster designs. Thrust levels indicated below may differ from OEXP requirements stated elsewhere in this report; such levels will be adjusted later.

Mars Transfer (1st & 2nd stage — 75-100k LOX/LH$_2$ engine. Currently no technology activities are focused on Mars transfer requirements for engines of this thrust class. However, much of the work being conducted in the CSTI ETO propulsion program is directly applicable, and some of the work on lower thrust engines being conducted in the chemical transfer propulsion Pathfinder element will also be applicable. Efforts directed toward high-thrust LOX/LH$_2$ engines are also part of the advanced launch system (ALS) technology program, the main emphasis of which is on low-cost manufacturing and production of expendable engines. In addition, consideration could be given to employing the Space Shuttle Main Engine (SSME) for this job. However, before determining what engines, whether existing or new, would be candidates for this application as well as what technology issues need to be addressed, both the total thrust level and the optimum thrust level per engine need to be established. Another issue is whether any or all of these stages could or should be recovered for reuse. Along with engine performance, the reuse issue will determine the amount of propellant each stage has to carry, which, of course, influences the number of heavy-lift launch vehicle (HLLV) launches

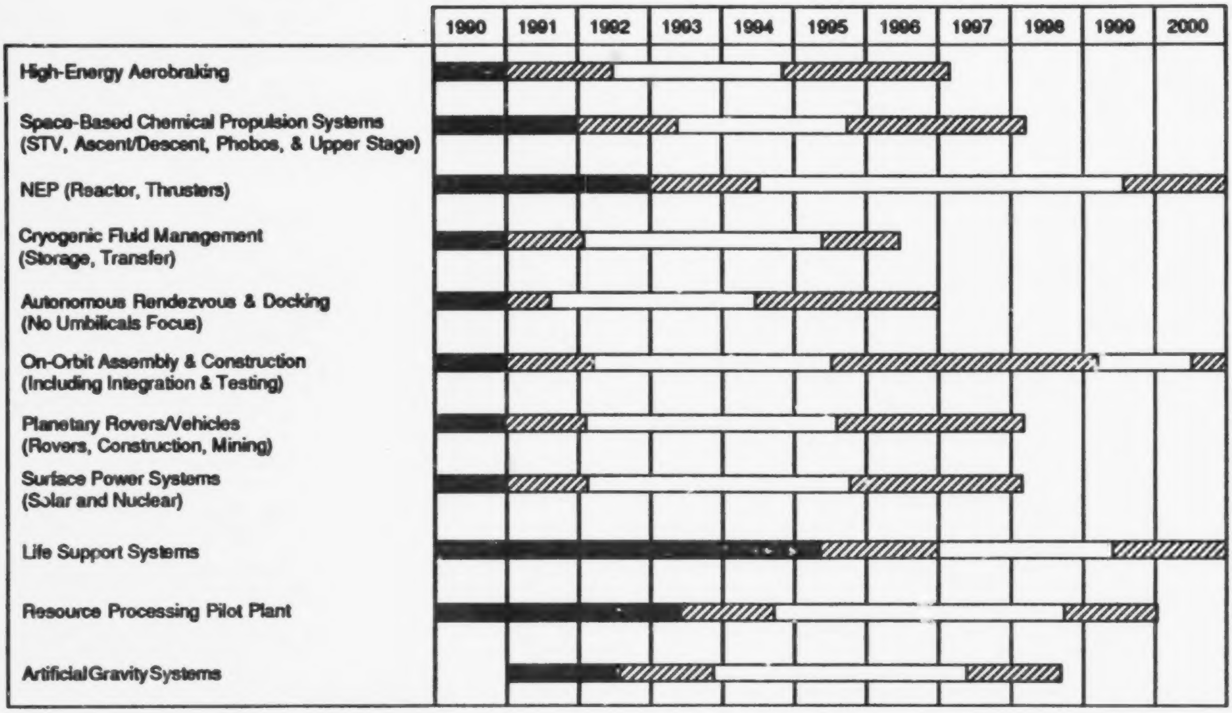


Figure 3.4.2-1.- Technology demonstrations accommodation summary schedule

that will be required to support the Mars mission. Currently, there are no plans to apply Pathfinder funds exclusively to high-thrust engine technology issues. The total thrust level requirements of the second stage may allow the use of clustered 20-40k engines; this tradeoff is being pursued in the CTP Pathfinder element.

Mars Transfer (3rd stage); Lunar Transfer; Mars/Lunar Return—20-40k LOX/LH$_2$ Engine. The technologies for this engine thrust class are the primary focus of the CTP Pathfinder element. They include high performance in order to minimize transfer vehicle propellants that must be delivered to the orbital assembly point by HLLV's extended service life (high design margins) for reusability, aeroassist compatibility (engine size), onboard condition monitoring for in-space maintainability, automated preflight servicing and checkout, fault-tolerant engine operations, and man-rated reliability and safety. The approach is to carry forward expander-cycle engine technology development into subsystem and breadboard engine system level technology validation and extension. For example, highly instrumented turbopumps will be assembled and operated over a wide range of conditions in order to accumulate a large experimental data base with which to validate design and analysis tools (analytical models/computer codes and advanced design concepts) previously developed in the R&T base program using laboratory and test rig equipment. An integrated component breadboard engine will then be assembled to develop analytical techniques for understanding and predicting component and subsystem interactions, as well as for predicting overall system performance and operational characteristics. Parallel technology efforts will be conducted, addressing (1) unique space-basing technology issues such as resistance to degradation from radiation and other in-space environmental conditions, diagnostic sensor development, the definition of integrated health monitor/control system architecture, and (2) the technology development of advanced control valves and actuators, expert systems, electronics for smart sensors, signal conditioning, engine transient models, etc. In addition, efforts will be focused on analytical techniques for defining internal dynamic loads, material behavior, and structural analyses aimed at predicting component life and improved material processes, fabrication techniques, and design configurations leading to longer life. Advanced components and subsystems will be designed and fabricated for installation and validation testing in an upgraded breadboard engine configuration.

Mars Transfer—High Power Electric Propulsion. Technology for electric propulsion systems for use in the Mars mission is being addressed in the cargo vehicle element of the Pathfinder program. The use of high power electric propulsion in the cargo stage of a split mission scenario for a Mars mission offers significant weight savings and a number of possible operational advantages, such as elimination of the need for aerobraking. There are, however, significant technical challenges associated with the use of electric propulsion in a vehicle the size of the Mars cargo vehicle. Power levels which will be >2 MW will likely dictate the use of a nuclear power source with power-to-weight ratios substantially better than those of the system being developed under the SP-100 program and the performance improvements expected from the NASA advanced conversion activities. Details concerning the power system technology are included later in this section. Two electric propulsion systems are likely candidates for this mission. Ion propulsion has undergone substantial technology development at lower power levels, and efficiencies near 75 percent have been demonstrated. However, the power per unit area of these thrusters is relatively low. As the diameter of the thrusters increases, reliability and efficiency decrease rapidly. The practical limit for the diameter of ion thrusters may be about 1 m. At the current power-per-area capability, a very large surface area (i.e., many thrusters) will be required. MPD thrusters offer a substantial increase in the power that can be processed per thruster. Relative to ion systems, MPD technology is in a state of infancy. Problems of efficiency and life must be solved if MPD systems are to be considered. Under the Pathfinder program, the basic mechanisms that control the life and efficiency of MPD thrusters and, if funding permits, the technology for increasing the power density of ion systems will be investigated. The unavailability of a facility which will permit high-power, steady-state testing of MPD's in a vacuum is expected to be a major problem in the timely development of the information required to assess MPD thrusters as an alternate electric propulsion system for the Mars cargo vehicle.

Mars and Moon Descent—20-40k LOX/LH$_2$ Engine. The critical technology need for descent engines is the ability to throttle over a wide thrust range (10/1 minimum) while maintaining a high performance level. Hovering and landing can consume large quantities of propellant, and high engine performance at the low thrust levels required for these maneuvers is essential for minimizing propellant consumption. In general, the technology being developed under the CTP Pathfinder element is applicable to both Mars transfer engines and Mars/lunar descent engines. However, technologies particularly unique to descent engines include high pressure engine operation (at full thrust), component design for efficient and reliable operation over wide flowrate ranges, and an engine control system that can provide the thrust variation required along with the necessary response while maintaining tight mixture ratio control. High combustion pressures at full thrust ensure high engine performance at reduced thrust (low combustion pressure), provided component and subsystem efficiencies can be maintained over wide operating ranges. Pump-fed LOX/LH$_2$ engines offer the most promise for achieving needed operational and performance goals.

Mars and Moon Ascent—20-40k (Undefined). Mars and lunar ascent propulsion systems must be capable of surviving hostile environments on the Mars/lunar surface for extended periods of time and provide reliable operations for the ascent maneuver when vehicle liftoff is scheduled. Although the LOX/LH$_2$ propellants may be storable on the lunar surface for extended periods of time (a concept that needs to be studied), it is highly unlikely that LH$_2$ could survive the martian surface environment without unacceptable propellant boiloff. Alternative choices for both requirements include storable propellants such as nitrogen tetroxide (N_2O_4)/monomethylhydrazine (MMH) or mildly cryogenic fuels such as methane (CH_4) or propane (C_3H_8) that could be used in conjunction with LOX. A compromise consideration could be LOX/MMH. However, no technology activities are currently underway addressing unique ascent propulsion requirements for engines of this thrust class. Specific technology issues include engine performance, heat transfer, cooling, health monitoring, and man-rated reliability. It is presumed that initial missions would use ascent propulsion systems transported from Earth to the martian and lunar surfaces. A later option could make use of propulsion systems designed to operate with propellants produced in situ at Mars or on the Moon. For example, oxygen (O_2) and carbon monoxide (CO) can be produced form the martian atmosphere, and O_2 and aluminum are abundantly available from lunar surfaces resources. Technologies for propulsion systems using these kinds of chemicals as propellants are being addressed in the space propulsion R&T base program. In any event, studies need to be conducted to establish firm ascent propulsion requirements and associated technology needs even for the more conventional propulsion systems, and appropriate technology programs should be initiated if required.

Power. The OAST advanced space power technology program includes the broad-based R&T base effort, the CSTI high-capacity power activity, and three relevant Pathfinder program efforts: SP-100 GES program support, surface power solar systems technology, and rover power technology development included in the planetary rover technology element. These activities, as well as a number of space power technology development cooperative efforts with the Department of Defense (DOD), are directed at providing the technology advancements essential to the conduct of projected OEXP missions. Advanced power generation, storage, and management systems technology, using both solar and nonsolar heat sources, comprise the primary thrust of these space power technology development endeavors. Emphasis is on demonstrating at the breadboard level compact, higher efficiency, reliable long-life systems for near-Earth and outer space operations. Space environmental interactions are a major consideration in the development of this advanced space power systems technology.

Unmanned Science Rover—0.5-1.0kW. The power technology advancement activity is a critical element of the Pathfinder planetary rover program, directed at developing the technology base for unmanned rover systems. The primary goal of the power system effort is to reduce mass to the lowest possible value, consistent with mission duty cycles and duration, and with the system's integration and installation requirements. With the assumption of a modular radioisotope thermal generator (RTG) as the prime power generation source, emphasis is being placed on the development of a SiGe/GAP thermoelectric material that can increase the RTG power conversion efficiency by a factor of two, thereby reducing its specific mass as well as the quantity of the very high-cost isotope fuel required for this mission. Another key technology objective is the development of the complimentary secondary battery systems with a specific energy density potential of 100 Wh/kg or greater. Advanced lithium cells will be designed and tested at represen-tative capacity requirements. An alternative high performance sodium-sulphur (NaS) battery system also will be investigated in this effort. Development of power-integrated circuits (PIC) or smart circuitry, a technology to enable integration of power and control circuits on the same substrate, is another task. Successful development of this technology could result in mass and volume requirement reductions as well as increased power system reliability. The objective of this PIC effort is to demonstrate the efficacy of a distributed power management and distribution system for rover applications. In the space power R&T base program, a novel electrochemical capacitor concept which could produce short-duration, high-power spikes will be assessed for possible application to rover system operations.

Manned Rovers—15-25kW. A utility base ("gas station concept") powered by nuclear or solar power could, via water electrolysis, produce the reactants needed for the primary fuel cells (PFC's) to generate electrical power for the manned rover system. The technology implications are the need for high efficiency fuel cell stacks, lightweight reactant storage system and fuel cell components, and thermal management systems. Regenerative fuel cell (RFC) energy storage technology being developed for the outpost solar power system will address the requirement for high conversion efficiency and some of the manned rover system's requirements for lightweight components. It may also be possible to satisfy this requirement with mobile solar power generation systems used in conjunction with a fuel cell storage system. High perfor-mance cell technology photovoltaic concentrators and deployable arrays technology, being developed in Pathfinder and the R&T base efforts, may be applicable to this requirement. Also, SP-100 reactor technology is scalable to this power level. However, shielding mass requirements and related operational constraints may make such a system excessively heavy and otherwise undesirable due to the resultant radiation environment. R&T base efforts addressing the technology issues of remote power transmission (satellite-to-surface) may provide a promising long-term electrical power alternative for manned rovers.

Utility Rovers—10-25kW. Based on the indicated requirements, the power system technology options addressed in the manned rover section are believed to be directly applicable to the utility rover.

Outpost—100-600kW. Two focused technology development programs directly addressing these requirements are ongoing CSTI high-capacity power and the proposed support of the Department of Energy's (DOE's) SP-100 GES program in Pathfinder. The objective of the SP-100 GES program is to develop and validate, by the mid-1990's, space nuclear reactor technology capable of generating 2.5 MW of thermal power (approximately 100 kW of electric power with thermoelectric conversion). The GES objectives will be achieved through development of selected technology components, by development testing and modeling, and finally by validating the performance of major subsystems assemblies (reactor and thermoelectric conversion) at representa-tive operating conditions in ground-test facilities simulating the operational environment.

The goal of the CSTI high-capacity power element is to develop the tech-nology for advanced power conversion and companion heat rejection subsystems that can be coupled to the SP-100 reactor and its heat transport systems. This would enable at least a fivefold increase in the electrical power that can be produced from the SP-100 reactor's thermal output (100 to more than 500 kWe), while more than doubling the power system's power-to-weight ratio. Improvements in efficiency and system growth potential, survivability, autonomy, reliability, systems life, mass reduction, and packaging are being sought. The nuclear reactor system can be designed to use its thermal output directly in cases where the operational need is for the direct application of this energy output to heat processing tasks of interest. This technology development activity addresses advanced high temperature and strength refractory alloys and composites; power conditioning, control, and distribution systems; power system self-diagnostics; and space environmental effects studies. In addition to the system growth potential that can be provided by the dynamic conversion system, the reactor technology being developed in the GES program has the ready potential for several-fold growth in thermal output without extensive new developments. Single-unit electric power delivery levels approaching 2 MW can be readily achieved using scaled-up SP-100 reactors and dynamic conversion systems. A deployable power plant is highly desirable for planetary exploration and is in need of study.

Initial Base (Sortie Mission). The Pathfinder program also includes an element directed at developing solar-based technologies for lunar and Mars surface operations in the 25-50 kWe power range. Solar systems will be needed to provide power during initial periods of operation while nuclear power systems are being placed in operation, or for the case of short-duration missions, such as Mars sortie flights, where the setting up of a nuclear power system may not be practical. These solar systems could provide emergency power in case of a major malfunction in the nuclear system. Therefore, the capability of a system to be stowed and deployed is being considered in the solar technology program. Emphasis is being placed on the advancement of technology in the key areas of high performance, lightweight energy storage systems, and extended-life, lightweight, high-efficiency power generation systems. Advanced hydrogen-oxygen RFC storage system components are being addressed in the energy storage area. In the power generation technology area, advanced photovoltaic thin cells and blanket interconnections are being addressed as well as very lightweight array designs capable of surface operation in a low-gravity environment. The development of an understanding of the effects of the lunar and Mars environment on components and systems performance and life is an essential element of the technology program. Because of the possible environment effects on photovoltaic systems, a top-level study of solar dynamic power generation is planned. Solar-based technologies provide a viable alternative approach at the lower end of the power level requirements and, by the application of modular units, could satisfy a potential 100 kWe surface operations requirement. The initial objectives of this effort are to demonstrate key components performance for the photovoltaic/RFC system and to identify the technical barriers for an alternative solar dynamic system.

Lunar base (with Resource Processing)—2-20MW. The nuclear power technology option addressed on the outpost section is believed to be directly applicable to a requirement of several megawatts of electric power through use of the growth options in conjunction with a multiple-unit utility power base scenario. There is direct program synergism with the CSTI high-capacity power and the Pathfinder SP-100 GES technology development programs discussed earlier.

Mars Transfer Vehicle Chemical Propulsion Stage—10-120kW. This requirement is ex-pected to be best met by application of solar power generation systems and electrochemical energy storage systems. Power technology issues being addressed in the R&T base and in the Pathfinder rover tasks and surface power elements are very similar. The main emphasis for this requirement is on respective duty cycles. If aerobreaking is utilized, arrays will be required to react and redeploy for the return trip.

Electric Propulsion Stage—2-10MW. Specific mass of the power system and power conditioning requirements to match the specific power demand cycles of the MPD or ion thrusters are the prime technology drivers for the nuclear power system for an electric propulsion stage. To be competitive, nuclear power systems for this application will need to demonstrate specific mass levels in the range of 5-15 kg/kW. The specific mass goal of the SP-100 GES program is 30 kg/kW; current GES system estimates are on the order of 40-50 kg/kW. In the OAST advanced conversion system technology program, dynamic conversion systems, lightweight radiator designs and materials, and lightweight heat pipes for the SP-100 reactor are being developed. These technologies are projected to yield a specific system mass of 20 kg/kW. To consider the SP-100-based power system for the electric propulsion stage application, additional mass reductions must be achieved. Advanced nuclear reactor power systems may need to be considered. The feasibility of future advances in solar power generation systems also may need to be assessed for this high-power, low-acceleration, long-duration application.

Artificial Gravity. Although the power level is still under study and no dedicated program is directed at artificial gravity, most of the technologies discussed above could potentially be applicable. There might be some specific requirements derived from the rotational motion of an artificial gravity system on the transmission of power across rotating joints and the control of liquids in some types of batteries and fuel cells.

Mars/Moon Descent and Ascent Vehicles; Lunar Transfer Vehicle. It is expected that the requirements for these applications will be covered by technologies developed for other applications. As the specific requirements surface, sections of the annual report will be augmented.

Life Support. NASA's life support technology program is composed of OAST efforts to develop physicochemical processes and systems to provide regenerative air revitalization and water reclamation technologies, and OSSA efforts to develop feasible biological processes and systems to provide regeneration of air, water, and food for long-duration human missions.

The OAST advanced life support program consists of a basic research and technology effort and the proposed Pathfinder effort on physicochemical closed-loop life support. The efforts are focused on developing chemical engineering technologies capable of providing regenerative life support functions, exclusive of food, for habitat applications, and regenerative and thermal control functions for portable life support systems (e.g., in space suits or human-operated surface vehicles). The technologies must exhibit acceptable weight and power consumption levels and provide for high system reliability and safety.

The base R&T program will focus on the development and validation of analytical simulation models of physiochemical processes, as well as analytical simulations of the component and subsystems. This analytical modeling capability will also lead to an integrated systems capability for analyzing complete life support systems. These models will provide analytical tools by which the feasibility of candidate processes can be screened before committing to further laboratory research and development, and by which proposed new processes might be evaluated. Candidate components and subsystems can be studied to perform trade studies in weight, power, performance, and cost within an integrated life support system model or within a spacecraft system model.

Knowledge-based and other artificial intelligence modeling techniques will also be used to develop automated process control methodologies for advanced life support systems. Adding control methodologies to these models will assist in the early identification of critical process parameters, systems interfaces, and sensors for monitoring and fault diagnosis. To enable monitoring and control of air and water quality, as well as automation and fault diagnosis for increased reliability and safety, substantial research will be initiated in specialized instrumentation and sensors. Real-time sampling, measurement, and analysis will be required to provide for crew health and safety at all times.

The Pathfinder portion of the advanced life support program will focus on the development of high-payoff chemical engineering processes and component technologies consistent with specific requirements being developed by OEXP for human space missions. Integrated life support systems models will be used to guide the selection and development of those technologies which will function most appropriately and efficiently in the context of established mission scenarios. The specific processes and technologies will include, in addition to air and water closure, trace gas and microbial contaminant monitoring and control for humans and plant systems, as appropriate.

Human Performance-Man/Machine Interface. The current OAST program in space human factors and the human performance element of Pathfinder deal with requirements for all four missions; however, the depth of coverage is not great and many specific problems are being addressed only in a generic manner.

Extravehicular Activity. The ongoing EVA program in OAST builds on previous Apollo and Space Station Freedom work. It will cover a general-purpose level of the technology requirements. As more specific issues arise, this program will provide an information base that can serve as a starting point for more focused efforts.

Aerobraking/Aerocapture. These requirements are currently covered by the CSTI aerodynamic flight experiment (AFE) program and the Pathfinder high-energy aerobraking program in OAST. The AFE program will be enabling for lunar and geosynchronous missions covering Earth reentry conditions with velocities on the order on 10 km/s down to altitudes of about 80 km. It will resolve current nonequilibrium radiative heating issues, establish performance characteristics of reusable (nonablating) thermal protection systems, provide a flight test of critical guidance, navigation, and control (GN&C) techniques, and be a valuable first step toward planetary return shock layer analysis capability and code validation. The high-energy aerobraking program will cover both piloted and robotic planetary missions and will encompass high-energy aerocapture (planetary and Earth), high-energy direct entry, and aeromaneuvering from orbital velocity (planetary) conditions. The critical and enabling disciplines to be covered will include mission studies/systems analysis, aerothermodynamics, GN&C, and materials and structures. The program will be planned and implemented in two broadly defined phases. The first will be a technology development phase resulting in the definition of selected aeroassist vehicle concepts and of a flight experiment. The second phase will be a technology demonstration phase culminating in a high-energy aerobraking flight experiment. Any additional studies that may be required will be defined at the completion of the program definition phase now underway.

Materials and Structures. Habitats for onorbit operations and surface operations are not being addressed in any significant way in either CSTI or Pathfinder; however, there is some related work in the R&T base. Similarly, innovative space structures are covered only in the R&T base. There is a need for a larger focused program in both of these areas. A better defini-tion of timelines, habitation periods, population size, and mix of structures (e.g., size and number) for focused applications would be helpful, but additional studies are not needed to begin a program in these areas.

Shielding materials are being addressed by two programs. A small effort is planned under the HIS thrust of Pathfinder to look at radiation shielding materials for transit spacecraft, surface habitats, and safe havens, as well as for suits and EVA protection. Characteristics of solar flares and cosmic radiation, including their biological effects, mission profiles, and length of exposure time will guide early study and analysis efforts. This could lead to a much larger program covering material development, if needed.

The issue of debris shielding is being addressed at a NASA-wide level through the Office of Space Flight (OSF), which chairs the Orbital Debris Steering Group (ODSG) responsible for advocacy and technical program development. Currently, the main focus of activity is protection on LEO, and a technical plan to address this problem has been developed. There is concern, however, that debris may also become a problem in GEO, especially for LEO/GEO staging and service operations for exploration missions. The issue of micrometeorites is not being emphasized, since it is a much lesser problem at LEO. If the natural environment away from the Earth (e.g., around Mars) poses a significant problem, this issue is probably best addressed through the ODSG.

Technology of cryogenic fluids tankage is covered under the NASA technology tasks in the ALS program and also as part of the National Aerospace Plane program. There is no dedicated materials program for artificial gravity tethers and booms. Other technologies related to these applications are covered in the **Management and Transfer of Cryogenic Fluids** paragraphs of this section.

Materials and structures technology for large scientific and communication systems is covered under several programs. The precision segmented reflector (PSR) element of CSTI is developing materials, structures, and controls technology for large astronomical instruments, such as the LDR proposed by OSSA. This CSTI activity has applications to precision structures in general, including communications. A smaller program funded under the base R&T focuses specifically on communication and scientific antennas. The significant technical difference is that PSR does not have an explicit optics activity; the antenna program has a strong electronics element but has less programmatic emphasis on advanced materials and structures. The precision required under PSR is at least an order of magnitude greater than for the antenna program, but much less than required for visible wavelengths. Control of flexible structures is also covered under CSTI. The program is focused on control/structure interaction (CSI), a unified multidisciplinary approach to designing, developing, and qualifying structures and controls for flexible spacecraft and platforms.

Surface construction methods are not covered in current OAST programs, but need to be addressed as a separate program. The methodology being developed under Pathfinder will exploit the microgravity working environment of onorbit operations but not zero gravity due to small gravity-gradient effects. Surface operations need to account for substantial gravity forces. In addition, precision construction for large scientific instruments would have greater emphasis. These two factors, gravity and precision, present the real challenge to surface construction. Also, robotic capability for surface construction operations would have to allow for the effects of the lunar surface chemistry and particulates on hardware.

Space construction is the focus of the in-space assembly and construction element of Pathfinder. The emphasis in the program is to develop technology to enable the onorbit construction of large, heavy-duty structures such as a Mars transfer vehicle, large fuel tank and habitat modules, and aerobrakes. The program includes mechanical joining, permanent joining and precise manipulation of large masses. However, it will also address to the extent possible utilities, inspection, precision construction, and heavy-duty deployables. The development of operational robotics capability and the definition of an in-space infrastructure, to accomplish all assembly and construction tasks, are integral and continuous parts of the program. By the end of FY 1992, basic robotic joining methodology/technology will be validated, and by the end of FY 1993, precision manipulation methodology will be validated. OAST will intentionally avoid construction technologies currently being emphasized by Space Station Freedom, including evolution and methods which will be made possible by current automation and robotics programs. However, the area of large-scale, heavy-duty assembly and construction is being addressed only in Pathfinder.

Controls. The objective of controls research and technology for advanced spacecraft, space platforms, and transportation vehicles is to enable control of large flexible space structures through development of modern control theories, advanced analysis, design/synthesis techniques, advanced failure recognition and reconfiguration algorithms, effective control system elements, new payload accommodations, and accurate experimental certifi-cation of system performance. The applications of controls are covered under the sections that address the specific applications (e.g., aerobraking and autonomous rendezvous and docking).

Automation and Robotics. The CSTI core robotics program is designed to provide leading edge technology in robotics, teleoperation, artificial intelligence, and human factors design, which will enable successively higher levels of autonomy in space remote manipulation. Specifically, the objectives of the program in robotics are to develop more advanced, versa-tile, and robust sensory and control capabilities. In teleoperation, the program provides electromechanical architectures and perceptual displays enabling highly dexterous multiarm control. In artificial intelligence the tasks apply to subdisciplines critical to robotics, such as task planning, monitoring, and failure diagnosis. The telerobotics focus of the space robotics program will have a sequence of technology integration and ground application demonstrations in order to provide technology for scene under-standing and its applications to robot spatial planning, control, and performance verification. The program incorporates planning and reasoning to provide technology for automated planning and verification of telerobot tasks including sequence design (procedural planning and scheduling), robot spatial planning, execution monitoring, and failure diagnosis, and to provide robot fine-motion for autonomous dexterous manipulation. In the control execution area, the program provides redundant manipulation for execution of complex tasks in automated and manual modes and in traded and shared control.

Autonomous Systems. The objectives of the program are to provide the basic technology in artificial intelligence required to achieve successively higher levels of autonomy in space operations and to adapt existing technology to aerospace use. The program seeks to establish research leadership roles in subdisciplines critical to space autonomy such as planning and scheduling, machine learning, cooperative knowledge-based systems, validation, symbolic multiprocessor architectures, and demonstrating the evolving capability of advanced autonomous control of systems.

The autonomous systems program will have a sequence of ground demonstrations to (1) ensure that the value of the component technologies will be tested in an integrated manner in mission operations environments, (2) permit periodic evaluations of the overall state of the art, (3) provide an objective method for determining the component technologies being developed, (4) provide a magnet for relevant component technologies being developed outside the program, (5) permit potential users to provide the program with feedback on its potential usefulness, and (6) serve as a testbed to validate artificial intelligence methodologies.

Autonomous Rendezvous and Docking. This element of Pathfinder will provide the capability for autonomous rendezvous and docking with the mission elements remaining in orbit for both piloted and robotic missions. Specific technologies to be pursued in this program include development of sensors and mechanisms, trajectory control requirements and techniques for operations in lunar and planetary orbits, and associated integrated guidance and navigation (GN&C) algorithms, such as automatic selection/execution/recovery techniques and multiple cooperative control. Sensors will be developed to provide long- and short-range tracking and relative navigation from several hundred kilometers down to the contact point of docking. Sensor technologies are driven by performance requirements for extended service life in hostile environments with long periods of dormancy. Docking mechanisms for both piloted and robotic vehicle operations will also be developed.

Autonomous Landers. The autonomous landing technology for lunar and Mars terrains will be developed and demonstrated in the Pathfinder program. The two major issues are precision landing at a preidentified location, and hazard avoidance during the final stages of landing (which includes real-time site selection during the landing process). The first problem is relatively close to the state of the art today and should be demonstrated by the third year of a fully funded program. The more challenging and more desirable technology area is the development of the capability to choose the final landing site in real time. To address this issue, significant advancements will be required in a number of related areas; these include real-time image processing, onboard computing, and sensors for hazard detection. Flight demonstrations in ground testbeds will provide a final validation of the technology.

Information Processing and Communications. The communications and computing technologies required to support exploration missions include the development of microwave and optical communications devices, as well as components for high-speed, special-purpose processors and high capacity data storage systems. The Pathfinder program has a dedicated element for the space demonstration of optical communications to develop the flight-qualified components and demonstrate the transfer of data at mega- to gigabits-per-second rates in Earth orbit and from deep space to Earth. The critical technology objectives include lightweight, highly efficient laser transmitters, high-precision pointing and tracking systems, large-aperture lightweight receiver telescopes, and high-sensitivity direct and heterodyne detection systems. In the CSTI program, the onboard processor architecture that provides reduction of outputs from high-rate imaging sensors is being developed and demonstrated. Optical disk recorders for high-rate scientific instrument processing are also being validated for information extraction, image correlation, and buffering of high-rate data streams.

Sensors and Instrumentation Systems. The purpose of the science sensor program is to develop the technology for detection in the submillimeter portion of the electromagnetic spectrum, to activate remote sensors using light detection and ranging (lidar) and differential absorption lidar (DIAL) techniques, and to support these technologies with the necessary passive and active cryogenic technology.

Detectors. The detectors element of the program will develop and demonstrate advanced detector and detector array systems with the requisite sensitivity, spectral coverage, reliability, and ruggedness for space flight. Maximum detective quantum efficiency and minimum cost are required, as are low-noise, large-array pixel formats and excellent imaging capabilities. The devices should cover a large dynamic range and photometric response over the integration times of interest. In general, development efforts will address all levels of performance, including responsivity, noise characteristics, spectral response, array size, power requirements/dissipation, and effects of space environments.

Submillimeter Components. Another goal of the program is to develop the technology for submillimeter sensing for space science to observe the cool clouds of interstellar dust collapsing to form planetary systems; to enable the ultraprecise measurements of spectral-line shapes from gaseous emission; to determine composition temperature, velocity structure, and dynamics for measuring species in the Earth's atmosphere in the submillimeter range; to understand the photochemistry of the stratosphere for stratospheric wind velocity measurement; and to measure pressure-broadened and Doppler-broadened spectral lines to allow precise determination of composition, altitude distributions, and general circulation patterns of the planetary atmospheres.

Lidar Sensors. The third objective is to develop and demonstrate advanced tunable solid-state lasers and gaseous laser technologies with quantitative technology targets to support the future Earth observation satellite (EOS). The requirements include continuous coverage of the electromagnetic spectrum from 25 to 10,000 nm; average laser powers in excess of 100 W (10 J per pulse at 10 Hz repetition frequency); electrical-to-optical conversion efficiencies in excess of 10 percent; narrow bandwidth operation of less than 1 picometer (10 m); laser lifetimes in excess of 10 laser firings (2 years' operation); modular lidar-transmitted technology for in-space servicing; and pulse durations less than 1 picosecond.

A parallel objective is to develop and demonstrate advanced electro-optical device technology in filters, modulators, wavemeter calibration systems, streak camera receiver technology, and wavelength control subsystems to provide lidar scientists with the capability to exercise "science-on-demand" with tunable lidar systems.

Cooler Systems. To develop and demonstrate advanced cryogenic systems with the requisite performance, ruggedness, and reliability for space flight is the final aspect of the sensor program. Maximum efficiency and minimum cost are required, as are extended life, excellent temperature stability, and the ability to change out and service instrument packages. The required levels of performance (heat loads at operating temperature, reliability and thermodynamic efficiency levels, weight and power constraints, storage or mechanical cooler lifetimes) far exceed the capabilities of the state of the art.

Rovers. The key objectives of the Pathfinder rover program are to develop and validate technology to enable the automated and piloted exploration of extensive areas of lunar and planetary surfaces. The initial focus is on automated martian rover technology for exploration and science. The key technologies for automated martian rovers are navigation, mobility, power, operations/autonomy, computation, architecture, and system integration. Development and integration of these technologies will allow orders-of-magnitude increase in the effectiveness of remote surface operations. Later technology needs are robust rover systems for automated construction and mining, and human-driven rovers for exploration. The generic technology requirements for manned and unmanned rovers are strongly related; the manned rover program element will be built on the technology base developed in the earlier unmanned rover program elements.

A planetary (including lunar) surface mobility capability is required to support planned future NASA missions identified in the 1987 NASA Space Goals Study. The MRSR project is the initial step in the manned Mars exploration program, and is the earliest NASA project identified as needing planetary sample return rover technology. MRSR is currently targeted for a 1998 launch, which requires technology readiness by 1992. Manned and unmanned rover technology to support exploration, mining, and construction functions is required for the manned lunar and Mars missions.

OAST has initiated the Pathfinder planetary rover program (PRP) to provide the required rover technology for enabling the manned and unmanned lunar and Mars programs, identified in the 1987 NASA Space Goals Study. Success in these future space programs requires that surface mobility technology be developed. The key technologies for automated martian rovers are navigation, mobility, power, operations/autonomy, computation, architecture, and system integration. Development and integration of these technologies will allow orders-of-magnitude increase in the effectiveness of remote surface operations. For example, it is impractical to have a martian rover teleoperated from Earth (i.e., one in which individual movements are controlled from Earth), because of the long signal time (30 minutes average round-trip). Each of the key required technologies is described in the following paragraphs.

Piloted Rover Technology. To maximize the usefulness of a manned lunar or Mars base, piloted surface transportation vehicles will be required. These vehicles would transport crews and instrumentation to sites not within walking distance of the home base. Most likely, a number of different types of vehicles will be required, from a short-range, man-in-life-support-suit, small-scale, lunar-type rover to a long-range, man-in-shirt-sleeve-environment, large-scale rover.

Autonomous Mining/Construction Rover Technology. A manned lunar or Mars base will require unmanned utility rovers (alleviating the need for manned surface EVA) to carry out operations such as mining and construction. The complexities and demands on the limited manpower of a lunar or Mars base will require significant autonomy to enable safe, efficient, and cost-effective mining/construction operations.

External Collaboration. Significant collaborative efforts with other agencies, industry, and universities are planned. The DOD, primarily through the Defense Advanced Research Projects Agency (DARPA), and the U.S. Army Tank Automotive Command (TACOM) are developing autonomous land vehicle technology. Significant synergy exists between autonomous rovers on Earth and those on planets other than the Earth. The results of the DOD efforts will be used in the NASA PRP. Moreover, it is the intent of the PRP to establish formal collaboration associations with the DOD program. The DOE has the national charter to develop nuclear power technology. As described in the power element of the program, a RTC technology development responsive to planetary rover requirements is necessary. OAST will establish a formal relationship with DOE wherein NASA can secure DOE support for development of planetary rover RTG technology.

It is the intent of the PRP to contract with industry for those technology products which industry can best provide. In the first couple of years of the PRP, industrial contracting is expected in the areas of mobility modeling and control, power component development, and data storage technology assessment. A collaborative relationship with Carnegie Mellon University (CMU) has been initiated. CMU will develop an innovative legged-locomotion Mars rover mobility prototype under a NASA grant which began in FY 1988 and is expected to continue through FY 1989 and 1990.

Resource Utilization. The resource processing pilot plant (RPPP) element of Pathfinder is addressing most of the areas identified in the OEXP PRD. Specifically, RPPP will validate the methodology for producing oxygen, metals, and building materials from lunar raw materials by the end of FY 1992 (including some effort in beneficiation/concentration). During FY 1993, the program will develop the basic design of a lunar pilot plant to validate this capability on the Moon and will begin building a laboratory testbed for technology development. Lunar mining will be considered but not heavily emphasized. Also, a small part of the program will be directed towards advanced concepts including resource processing on Mars. The thrust of any martian processing will be directed towards water and oxygen production. There is a need for a better definition of the benefits of martian resource processing, including timelines and economies of scale.

Surface Science. All of the areas for unmanned mission are being addressed under the sample acquisition, analysis, and preservation (SAAP) element of Pathfinder. By the end of FY 1992 basic technology sould be developed in all areas. However, with the current projected level of funding, both detailed concept development and full autonomy will not likely be developed for all areas. Decisions on whether to emphasize automation or design will be made early in the program for each area. A conceptual design of an SAAP system compatible with an unmanned Mars mission will be designed by the end of FY 1993, and subsystems for a laboratory testbed will be demonstrated. Many very relevant technology areas for surface science are covered in the preceding **Information Processing and Communication and Sensors and Instrumentation Systems** paragraphs.

Management and Transfer of Cryogenic Fluids. The ongoing OAST cryogenic fluid management program in the R&T base program addresses all the OEXP technology requirements in the PRD. This same program will become part of the onorbit cryogenic fluid depot line item under Pathfinder in 1989 — if Pathfinder is approved in its totality.

The overall objective of the cryogenic fluid management base R&T program is to develop the technology required for the storage, supply, handling, and transfer of subcritical cryogenic liquids in the low-gravity space environment. The approach used to achieve this objective began with a thorough identification of technology requirements. This task was originally per-formed by the NASA In-Space Cryogenic Fluid Management R&T Planning Committee in 1979 and subsequently revisted by a NASA Marshall Space Flight Center (MSFC) contract with General Dynamics, by an Air Force Rocket Propulsion Laboratory (AFRPL) contract with Martin Marietta, and most recently by the user community consisting of the NASA centers, DOD, academia, and industry.

The program focuses on developing analytical models describing the various governing processes, performing a series of ground-based experiments, and formulating in-space experimentation. Ground-based experiments will provide limited validation of the analytical models, investigate required technologies which do not need the low-gravity space environment, and provide data to be added to the developing data base.

Analytical model development describing the important physical processes has been underway for several years. This effort has involed academia, other Government agencies, industry, Lewis Research Center (LeRC), and other NASA centers. Typically, this effort has involved and will continue to involve 1) stand-alone models developed in-house at LeRC from basic principles, 2) complementary modeling supported by grants, memorandums of agreement, and contracts, and 3) an integrated modeling effort in which the basic modeling is combined with the out-of-house complementary modeling to create an integrated, user-friendly computer code (CRYOTRAN) to predict fluid and thermal behaviort of cryogenic systems in the low-gravity space environment. The analytical model will be documented and disseminated to the user community. These computer codes, once validated, will be used to aid the designers of operational in-space cyrogenic systems.

Ground-based experimentation will be performed at LeRC, at university and contractor sites under grants and contracts, and potentially at other NASA centers. These experiments are expected to range from fundamental process evaluations to system-level experimentation. They will include fundamental studies of interfacial heat and mass transfer, Joule-Thomson expander characterization, chilldown and nonvented tank filling, tank pressure control, thermal stratification, liquid sloshing, insulation system evaluation, quanity/mass gauging and testing, fluid mixing, pressurization (both autogenous and noncondensible), thermal subcooling, subcooled or slush hydrogen transfer, mass flowmetering, screen acquisition degradation, and cryogenic liquids, nitrogen, and hydrogen, as well as other fluids such as water and certain refrigerants. Space experiments are necessary to generate additional data for those processes and technologies requiring the low gravity environment for development.

Artificial Gravity Spacecraft. Some activity will be initiated under the HiS thrust of Pathfinder to look at rotating spacecraft (vehicles) that provide artificial gravity. This would involve systems engineering and structural analysis studies including extension-retraction concepts; the dynamics, stability, and control of spin-up, spin-down, and steady operations; and identification of technology requirements.

Construction of rotating systems may later be addressed under the in-space assembly and construction element of Pathfinder but this would be to identify issues and requirements within the general context of construction methods and space-based infrastructure. A program focusing on artificial gravity spacecraft will be needed to address all the configuration, structure, mechanism, and control issues adequately. It could be a separate program or a part of a more comprehensive one on large-space-structure concepts that could also include space habitats and their operational requirements.

Surface Launch Systems. This area is covered generically under the propulsion tasks, but no focused program has been created to meet either lunar or martian requirements.

3.4.4 Support Required from Other NASA Organizations

The technology requirements of other NASA offices to meet their supporting objectives for OEXP have not been formally established; however, to some degree they are implicit in the specific requirements stated for the other NASA offices in OEXP's PRD. In general terms they are covered in the technology programs described above. For example, the transportation systems that need to be developed by OSF are covered in generic terms by the OAST propulsion and materials and structures programs. The technologies required by second- and third-generation space stations are also under joint study. In the science and precursor area the Pathfinder and CSTI programs address the technology needs as defined to date. The potential of combining a precursor Mars sample return mission with a demonstration of the technologies needed for a manned Mars mission has been identified. As the other NASA offices continue to develop and refine the understanding of future systems that will be needed for human exploration of the solar system, the resulting technology requirements and opportunities need to be formally established.

3.4.5 Conclusions

The current R&T base and focused technology programs cover most of the identical needs for the human exploration of the solar system. However, the level of coverage could be substantially increased to add much-needed depth to the technology options. A broader program is also needed to cover the new questions that will surface as lunar and Mars human exploration missions are studied in more depth. The most urgent need for identifying technology opportunities and requirements is a clarification of the potential future scenarios. The more specific the scenario, the clearer the prioritization can be for the technology program. One other top-level need is an exploration study library, starting with bibliographies and hard copies of all known existing studies. As future studies are documented, copies should be maintained in at least one centralized place to create an exploration library.

Some of the critial questions and issues raised by the technologists in planning the Pathfinder program are listed for reference below.

Propulsion. What are the projected thrust levels, total impulses, and payload requirements for future missions to the Moon and Mars; and what is the optimum set of transportation vehicles to perform these missions? What would be the expected staytimes at both the Moon and Mars? What are the requirements for restart? What is the role of low-thrust propulsion for cargo vehicles? What performance levels would low-thrust transportation systems have to demonstrate in order to be viable candidates for exploration missions? What specific impulse, total impluse, thrust-per-kilogram, and payload capability would be required? What refueling strategies are dictated by mission scenario strategies? What is the potential utility of the Nerva-class propulsion system being revitalized by the DOD, and if it does not satisfy new requirements, what can OAST do to build on the existing technology base?

Power. What are the power tradoffs for surface power systems during the manned visit, outpost, and the base stages of lunar and Mars exploration? What are the benefits to the science objectives, mission objectives, and mission viability of having various levels of power? What are the expected durations of missions? What is the frequency of missions and the degree to which specific power systems can potentially be used for several applications (i.e., lunar, Mars, and in-space)? What are the power requirenments for life support on lunar bases and Mars transfer vehicles? What are the power requirements for surface rovers, both manned and unmanned (specifically for mobility in computing, for drilling, and for communications)?

Aerobraking. A detailed analysis of the capture parameters at Mars and at Earth from both the Moon and Mars is needed. Included in the results should be the pointing accuracy, the velocity range, and the range of capture orbits. Analysis of Viking experience in the Mars atmosphere is particularly important.

Humans in Space. A representative functional scenario should depict the first human surface presence establishing a lunar base, including site exploration, outpost establishment, and evolution to a multipurpose base. Each stage should show what the human is expected to do, both functionally and operationally. Based on these scenarios, hardware needs leading to technology requirements and opportunities should be developed.

For the first piloted mission to Mars, a representative human function and activity profile should be created. It would include the stages of preparation in Earth orbit, activities enroute, the Mars encounter, and return and rearrival activities. From these human performance expectations, hardware and technology requirements can be derived.

A study of configuration options for a piloted Mars vehicle should be conducted to provide alternative structural arrangements that provide high-Z particle radiation shielding in zero-gravity and artificial-gravity conditions. This information would help guide engineering studies of rotating systems and radiation-shielding materials studies and clarify shielding mass penalties.

There is also a need to develop a bibliography of past lunar base studies that can be used as a reference for both human performance and human factors technology programs.

In-Space Assembly and Construction. A comprehensive study is needed to identify the major structural systems required for a space infrastructure to support long-range missions, including onorbit, vehicular, and surface operations structures. Estimates are needed of size and configuration, operating environment, external loading conditions, estimated longevity and reuse, general repair and refur-bishment requirements, structural system performance requirements (e.g., maintain micro-g or artificial 1-g conditions). Systems and requirements need to be prioritized with regard to mission application and timeframe. The aerospace community's current key technological strengths and weaknesses regarding in-space assembly and construction of major structural systems need to be identified and key problems need to be solved. The product of this study, which OAST will use to identify the most critical technology areas, will be a clearer definition of technology needed to enable future missions and of its completion schedule. The study will also provide a cross-check of the current state of technical readiness as perceived today.

Automation and Robotics. One of the most critical issues related to the human exploration of the solar system is the optimum use of automation and robotics. Beginning with the precursor missions, what is a reasonable evolution of robotic task capability over the next 20 years? A preliminary projection is needed to begin mission planning and to define the role of humans. Similarly, what is a reasonable expectation for the automation of spacecraft and transportation systems? Trade-offs of various robotic and human teams need to be studied in various mission scenarios to define the critical technologies to be emphasized in the supporting technology programs.

Fault-Tolerant Systems. The following studies are needed on photonic systems that will provide the maximum benefit to several detailed requirements:

a. Develop information management system requirements for a human transfer vehicle to Mars that is fault-tolerant and has a high enough data-rate capacity for the collection and communication of data for vehicle and scientific instrument control, and enough video data communication capability to make an astronaut comfortable. Several critical questions that should be answered include: What areas of photonic technology will enhance and enable the fulfillment of these requirements (for instance, photonic fiber-optic networks with photonic switching nodes)? What is the comparison between photonics and electronics in the spacecraft environment (radiation resistance, power, volume weight consumption)?

b. Review all of current photonic technologies, secure and nonsecure, to find all pattern recognition and multispectral processors that (1) have potential to fulfull a planetary rover's vision requirement for navigating natural, rocky terrain while avoiding >1-m-diameter rocks, and(2) for scientific sample recognition, have potential to process vision information fast enough to guide an autonomous lander around rocks >1 m in diameter during the last few minutes of flight. Compare these with similar digital image processing systems, taking into account the power, weight, and volume requirements.

c. Conduct research to find out the in situ, fiber-optic/integrated optical, fault-tolerant sensor development needed for interplanetary transfer vehicles with people onboard. Potential sensing needs are for sensitive strain sensors to be embedded in composite skins, high-temperature sensors for planetary liftoff to return to Earth, etc.

Lunar/Mars Communication Networks. Estimates of requirements are needed (i.e., rate, utilization, frequency, ground requirements). What types of lunar and Mars orbits would be used? How would these specialized communi-cations satellites interface with the current tracking and data relay satellite systems?

Orbital Geological Survey. What image resolution and spectral capability is required at the Moon and at Mars for site selection and surface operations support during robotic and manned missions? How much image processing will be performed on the satellite?

Science. What specific sensor and supporting technologies need to be developed in support of science objectives on the Moon and on Mars? These of course will be derived directly from the definition of the companion science objective associated with human exploration.

More thorough and explicit identification of requirements in the areas indicated above will provide technologists with the level of detail necessary to ensure the greatest relevance to the human exploration scenarios.

3.5 EXPLORATION IMPACTS TO COMMUNICATIONS, TRACKING, AND SATELLITE SUPPORT SYSTEMS

3.5.1 Role of the Office of Space Operations (OSO) in Manned Exploration

The NASA Office of Exploration (OEXP) has asked OSO (code T) to assess the four case studies on human exploration of the solar system currently under development by OEXP and to estimate OSO-related system needs, technology requirements, and design options of such a program. The Jet Propulsion Laboratory (JPL) and Goddard Space Flight Center (GSFC) jointly are tasked to carry out this assessment. This section reports on the joint JPL/GSFC assessment activities in the four OEXP exploration scenarios:

a. Human Expedition to Phobos

b. Human Expeditions to Mars

c. Lunar Observatory

d. Lunar Outpost to Early Mars Evolution

The OSO responsibilities include telecommunications, navigation, and information management (TNIM) systems design. This FY 1988 edition of OSO studies emphasizes the telecommunications design principally because navigation and information management system requirements have not yet been identified. Telecommunications link analysis must include all elements of the end-to-end system. As studies progress, detailed spacecraft telecommunications designs may be developed elsewhere; however, the TNIM system design is expected to remain central to the overall OSO responsibility.

Point OSO designs are developed for each case study, telecommunications requirements are assumed, current-versus-projected support capabilities are compared, support technologies are identified, and plans for continued assessment activities are described. The objective is to bracket the needs with a series of design options. Current organizational and technical issues are also raised. This preliminary assessment documents work accomplished in FY 1988.

3.5.1.1 OSO Study Team Goals

The OSO goals begin with a broad generic architecture study addressing telemetry, video, command, navigation, data processing, access, and communications networking services needed to support OEXP human exploration scenarios. As the specific requirements for human exploration become firm, the effort will focus more on the detailed infrastructure required by OSO to meet these needs. Specific near-term goals are listed.

a. Establish a set of validated requirements using a mission-OSO-interactive design team environment.

b. Develop appropriate telecommunications, navigation, data-handling, and operational concepts.

c. Recommend balanced mission-OSO design evolution with alternatives, based on life-cycle cost factors.

d. Identify needed technology and risks for OSO functions/systems.

e. Recommend and coordinate flight-related technology development.

f. Support technology demonstrations prior to commitment.

3.5.1.2 TNIM Support Options

A number of options exist for the architecture of TNIM support systems. The choice of options to pursue will depend on a number of factors, including the selected exploration scenario and its mission need, the overall cost-effectiveness of the resulting architecture, NASA life-cycle costs, and the support needs of current unmanned NASA and international cooperative missions at remote bodies. A final selection of the human exploration scenario to proceed upon is planned for the early 1990's; therefore, all candidate support options must be identified and analyzed early and the implications of each understood for each exploration scenario. The current list of OSO configuration support options includes

a. Present OSO services using Earth terminals support:
   1. Telecommunications, radiometric, and navigation of deep-space missions with the mission supplying in situ relays and navigation
   2. Near-Earth missions with telecomm and navigation using tracking and data relay satellite systems (TDRSS)
   3. Mission data handling

b. Present OSO services with addition of dedicated manned mission support network

c. Present OSO services with addition of OSO-managed in situ telecommunications and navigation networking, and data-handling services at remote bodies. This option would be driven by economies of scale resulting from support of many in situ spacecraft/landers.

3.5.2 Case Study Needs Assessment

3.5.2.1 Generic Telecommunications Requirements

Seven generic data types have been defined which span the spectrum of manned missions telecommunications requirements: video, voice, science, engineer-ing, telerobotics, command, and data load. The requirements for and definitions of these data types are summarized in table 3.5.2-I and described in more detail below.

TABLE 3.5.2-I.- DATA TYPES AND DATA RATE REQUIREMENTS

Data type	Data rate (mb/s)	Comments
High-rate video	100	● Single channel, color, 512 x 512 pixels, 8 bits/pixel, 30 frames/sec
Low-rate video	0.20 (0.1 fr/sec)	● Single channel, monochrome, 512 x 512 pixels, 8 bits/pixel, 0.1 frames/sec
Voice	0.02	● Single channel, links with manned vehicle
Science telemetry	to 10 to 300	● Low duty cycle spectral scanning or SAR imaging with data storage available ● No data storage, spectral scanning or SAR imaging
Engineering	0.2 0.002	● Per manned spacecraft ● Per unmanned spacecraft
Telerobotics	0.2 200	● Command channel, per rover ● Stereo video, color, 512 x 512 pixels, 8 bits/pixel, 30 frames/sec
Command	to 0.002	● Per spacecraft or science platform/site
Data load	to 1.0	● Earth to manned vehicles or manned outposts, access to Earth data bases

Notes:

1. Data rates listed above are maximum raw data rates per data type. Total data requirements per link depend on the number of data types required per link, data compression techniques employed and the schemes used for multiplexing intermittent data streams onto a single link.

2. Data storage capacity on high-rate science instruments allows reduction of required data rate by allowing intermittent high-rate data to be transmitted over a longer period.

3. 2-way high-rate video channels and voice channels are required between Earth and manned vehicles and outposts where humans will be for extended stays. This is primarily for personal communications with family/friends, and for news and entertainment.

Video. The proposed manned missions to Mars and the Moon involve humans in transit to/from or inhabiting a remote base station for durations up to 45 months. For the sociological and psychological benefit of the men and women on such extended missions it is very desirable to provide two-way video/voice links to mission operations personnel, relatives, and friends, and to provide a selection of video-format training, entertainment, and news. Current technology requires a data rate of 100 Mb/s to support a high-quality uncompressed digital color video channel. Expected data compression technology developments may reduce these needs by better than a factor of ten. It is assumed that access to one or more of these high-rate video channels would be required for each transit vehicle and for each remote site involving extended manned occupation. High-rate video links between manned vehicles and manned bases may also be desirable at times. In addition, at appropriate times the high-rate video capability on the return link to Earth would allow the American public to participate vicariously in mission activities as a "real-time" observer.

A low-rate video type, 0.1 frames/s, would support imaging for science and technical information transfer not requiring the real-time characteristic of high-rate video. Compressed image data similar to that currently used on the Voyager mission could be relayed at a net relay link data rate of 200 kb/s.

Voice. A high-quality voice link is required between any two manned locations, or to Earth for the purpose of coordinating mission activities and information transfer. A high quality delta-modulation-coded digital voice channel requires, at present, a 40-kb/s data rate channel. Advanced voice coding techniques may reduce this data rate by at least a factor of two. The long-turn-around light time to Mars precludes normal two-way conversations. Instead, voice communications will tend to be drawn-out and well-considered information transfers, but still containing the necessary verbal psychological messages.

Science Data. Science data include radio science, meteorological, geophysical, multispectral scanning, or SAR imaging data produced by any number of orbiting or landed science instruments. Data rates can range from an average of 10 to 100 b/s for geophysical monitoring instruments up to 300 Mb/s for raw SAR imaging data.

Engineering Performance Data. Engineering data include monitor and health data for spacecraft, human habitats, surface vehicles, and any other needed equipment monitoring. A peak rate of 2 kb/s is expected for unmanned spacecraft, consistent with the Voyager spacecraft peak engineering-data rate of 1200 b/s. A peak rate of 200 kb/s is expected for manned spacecraft, consistent with peak engineering-data rates of 50 kb/s and 130 kb/s employed on Apollo and the Shuttle respectively. Engineering data rates for habitats and other equipment need to be developed.

Telerobotics. The unmanned rovers proposed for both lunar and martian missions will require, at a minimum, stereo video from the rover to the human control console. Two digital video channels may require a data rate of 200 Mb/s. Additional data at a rate to be determined will be required if simulated tactile/motor feedback to the controller is desired. The operator-to-rover link will require 200 kb/s per rover for control of locomotion and robotic functions.

Command. The command-data type involves uplink of low-rate information, command and control data to spacecraft and science platforms/sites for the purpose of remotely staging operations and device sequencing. A peak command-data rate of 2.0 kb/s is expected, consistent with the 2.0 kb/s used on the Shuttle cargo command interface.

Data Load. The extended-duration manned expeditions, experimentation, and mining operations proposed will require access to Earth data bases for science data interpretation, instrument and facilities troubleshooting/repair, updated or modified artificial intelligence (AI) strategy matrices/data, and effective logistical planning. A data rate of 1.0 Mb/s is considered adequate for rapid upload of this information to manned spacecraft or outposts.

How the specific data types combine into a total telecommunications capacity requirement depends on the number, duty cycle, and scheduling of individual links. The next section will address the first of these issues: connectivity requirements between the Earth, spacecraft, science instruments, etc., per mission case study. Future studies will address the next level of detail.

3.5.2.2 Mars-Related Case Studies

Figures 3.5.2-1 through 3.5.2-5 describe the link requirements for the manned Mars and lunar mission options. Each figure contains a pictorial presentation of the nodes that require communications. A link required between two nodes is indicated by a solid interconnecting line. Each node is assigned a single letter code which is used in the accompanying table as a symbol for link connectivity. For example, in figure 3.5.2-1, AB denotes the link between the Earth (A) and the manned vehicle (B). ADB denotes the same link via a relay vehicle (D). Data types required on a particular link are indicated by an X in the column corresponding to the specific data type. A dash (-) indicates that a particular data type is not used.

Human Expedition To Phobos (sprint to be there first). The communication architecture for the Human Expedi-

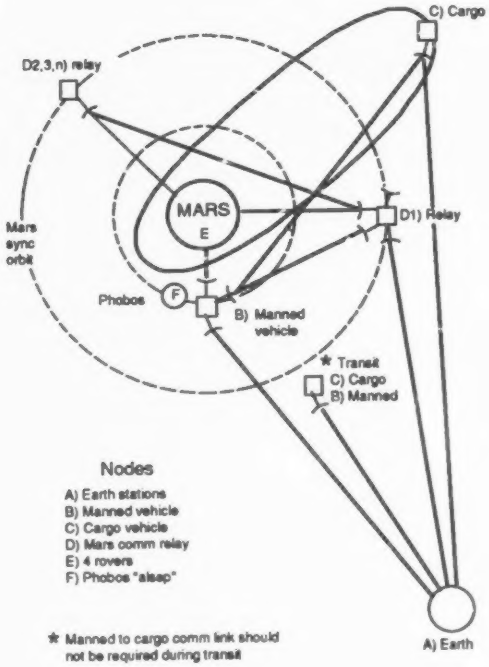


Table of data types:
connectivity per link

Data type / Link	Video high rate	Video low rate	Voice	Tele-robotics	Sci. data	Eng. telm	Cmd, data load
AB, ADB	x 2way	x 2way	x 2way	–	x	x	x
AC BC	–	–	–	–	–	x	x
BE, BDE	–	x	–	x	x	x	x
D1D2, 3	x - - - - - Relay purposes only - - -	x	x	x	x	x	x
EDA	–	x	–	–	x	x	x
FDA	–	–	–	–	x	x	x

x – means data link is required. Required link availability is nominally continuous coverage = 0.98. This does not include occultation outages which are tbd, depending on specific link geometries. Availability is not utilization.

Figure 3.5.2-1.- Human Expedition to Phobos: comm links.

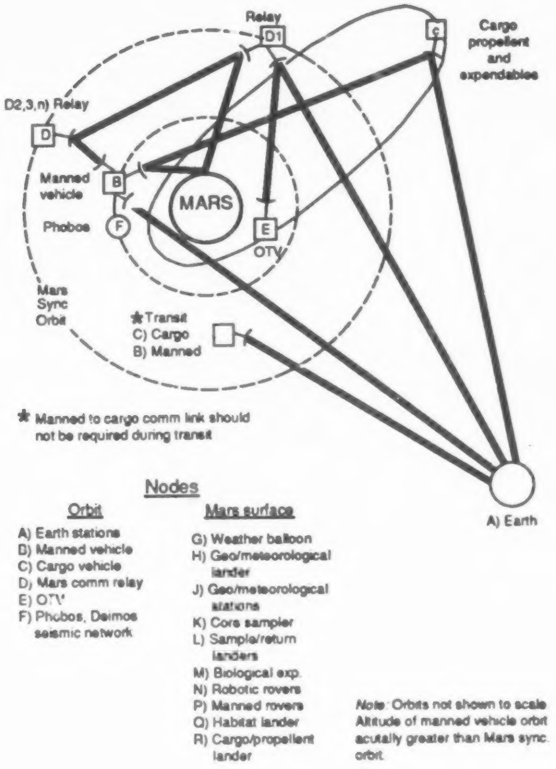
tion to Phobos scenario is shown in figure 3.5.2-1. This case study had the requirement for two-way continuous voice, video, and data communication with the manned vehicle throughout all phases of the mission and a continuous telemetry link with the cargo vehicle until the manned vehicle is inserted into its Earth-return trajectory. During rover operations on the planet surface, there is a requirement for relay of low-rate video and science data to Earth. Mars local communications require a high-rate video and command link for telerobotic operations of the rover from the manned vehicle approaching Mars or near Phobos.

The 100 to 200 Mb/s data (uncompressed) required to support continuous high-rate video, voice, and science data dominate the data rate requirements on the Mars-to-Earth link. Similarly, high-rate telerobotics video and control data dominate the data rate requirements of Mars local communications. A combination of improved data compression technique and higher-data-rate telecomm flight hardware will be needed to support these data types.

The need to minimize or eliminate link outages due to occultations suggests the use of one or more communication relay satellites in Mars orbit to accommodate continuous communications between the manned vehicle, the surface rover, and Earth. The short ranges involved in Mars local communications and the thin martian atmosphere allow the effective use of Ka-band (32 GHz), W-band (90 GHz), or optical frequencies on relay satellites to support up to 1 Gb/s data rate on surface-to-surface, surface-to-orbit, and orbit-to-orbit links. Selection of specific frequencies will depend on technology readiness, costs, maximum data rate needs, Mars surface coverage requirements, and considerations of the impact of martian dust storms on the communications link performance.

Human Expeditions to Mars (Sprint to Be There First). The communications architecture for the Human Expeditions to Mars scenario is shown in figure 3.5.2-2. All the requirements of the Phobos case study described in the previous section apply to this case study. Two areas require additional telecommunications capacity over the Phobos mission. First, the presence of humans on the Mars surface extends the coverage of the two-way continuous voice, video, and data links with Earth to include the Mars surface. Additional video and voice links between the Mars surface base and remote surface vehicles, crew, and equipment are required. Second, an increased number of dispersed Mars surface science instruments will require increased coverage and capacity for relay of data to Earth over the experiment lifetime.

Again, the 100 to 200 Mb/s of high-rate video and science data dominate the capacity requirements on the Mars-Earth link and on Mars orbit-to-orbit, orbit-to-surface, and surface-to-surface links. Specific consideration will be given to the development of a satellite relay system that can support a diverse spectrum of spacecraft types and landers, manned and unmanned. The support of minimum complexity science instruments is desirable to encourage a maximum number of low-cost U.S. and


Relay
D1
C
Cargo propellent and expendables
D2,3,n) Relay
D
Manned vehicle
B
MARS
Phobos
F
E
OTV
Mars Sync Orbit
★ Transit
C) Cargo
B) Manned
★ Manned to cargo comm link should not be required during transit
A) Earth
Nodes
Orbit
A) Earth stations
B) Manned vehicle
C) Cargo vehicle
D) Mars comm relay
E) OTV
F) Phobos, Deimos seismic network
Mars surface
G) Weather balloon
H) Geo/meteorological lander
J) Geo/meteorological stations
K) Core sampler
L) Sample/return landers
M) Biological exp.
N) Robotic rovers
P) Manned rovers
Q) Habitat lander
R) Cargo/propellent lander

Note: Orbits not shown to scale. Altitude of manned vehicle orbit acutally greater than Mars sync. orbit.

Table of data types: connectivity per link

Data type / Link	Video high rate	Video low rate	Voice	Tele-robo-tics	Sci. data	Eng. telm	Cmd, data load
AB, ADB	x 2way	x 2way	x 2way	–	x	x	x
AC BC	–	–	–	–	–	x	x
AQ, ADQ BQ, BDQ	x 2way	x 2way	x 2way	–	x	x	x
AE, ADE BE, BDE QE, QDE	x 2way	x 2way	x 2way	–	–	x	–
BL, BDL	–	–	–	–	–	x	x
BN, BDN QN, QDN	x	x	–	x	x	x	x
BP, BDP QP, QDP	x 2way	x 2way	x 2way	–	x	x	–
D1D2, 3	x - - - - - Relay purposes only - - - - -	x	x	x	x	x	x
(G, H, J K, L, M) TO DA	–	–	–	–	x	x	x

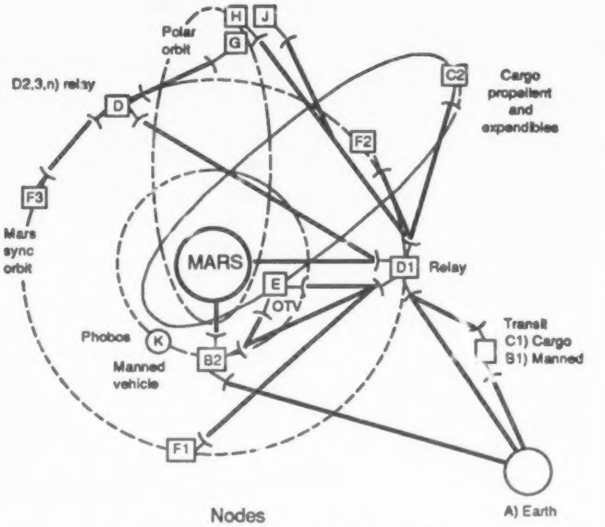
x Means data link is required. Required link availability is nominally continuous coverage = 0.98. This does not include occultation outages which are tbd, depending on specific link geometries. Availability is not utilization.

Figure 3.5.2-2.- Human expeditions to Mars: comm links.

international cooperative science spacecraft and probes that could use the U.S. Mars communication network to relay science data back to Earth. The exact number of relay satellites and the extent of their onboard capabilities for data compression, data storage, encoding, and switching are to be determined, pending link analysis, loading estimates, and occultation outage predictions. For all case studies the desire is to provide adequate and reliable communications services. Communications system design should result in performance that degrades gracefully with single device failures and ultimately fails last in reference to the other flight and support systems.

Lunar Outpost-to-Early Mars Evolution (Mars Portion). The telecommunications requirements for the Mars portion of the evolutionary case study will be similar to the humans-to-Mars case study with two or three additions, as presented in figure 3.5.2-3. Science instruments deployed in orbit such as meteorological, geoscience, and imaging cartographic satellites suggest the use of TDRSS-type data link consolidation for relay of information back to Earth at data rates up to 300 Mb/s; this rate may be reduced significantly through the availability of efficient data compression techniques. Link geometries for the evolutionary case study are sufficiently complex to recommend two or more capable relay satellites. As in the Mars sprint case study, support of minimum complexity science instruments and international cooperative spacecraft is desirable. Mining operations on the Mars surface and on Phobos will require additional communications links to fixed mining sites and to propellant cargo transport vehicles. Specific communications requirements in support of mining operations are to be determined. Additionally, the evolutionary case study facilities must support repeated precision landings (±3 m) to the same site.

Guidance and navigation issues have not been addressed to date in this analysis, but the evolutionary development of a Global Positioning System (GPS) or interferometer-type navigation network tied to the martian system is desirable to provide incoming spacecraft with a highly accurate Mars-centered surface and orbital navigation reference. Implementation of communications/navigation capabilities will be evolutionary in nature, corresponding to the evolution of requirements inherent in this case study strategy. The intent is to make a long-term investment in the Mars communications/navigation network that will pay off over multiple missions



Nodes

Orbit

A) Earth stations
B) Manned vehicle
C) Cargo vehicle
D) Mars comm relay
E) OTV's
F) Meteorological sats
G) Polar meteorological sat
H) Particle/fields sat
J) Upper atmosphere expl sat
K) Phobos seismic network and propellent mining facility

Mars surface

L) Penetrators
M) Weather balloon
N) Geo/meteorological stations
P) Core sampler
Q) Sample/return landers
R) Biological exp.
S) Robotic rovers
T) Manned rovers
U) Lander habitat
V) Lander propellent

Table of data types: connectivity per link

Data type LINK	Video high rate	Video low rate	Voice	Tele-robotics	Sci. data	Eng. telm	Cmd, data load
AB, ADB	x 2way	x 2way	x 2way	–	x	x	x
AC,ADC BC,BDC	–	–	–	–	–	x	x
AU, ADU BU, BDU	x 2way	x 2way	x 2way	–	x	x	x
AE, ADE BE, BDE UE, UDE	x 2way	x 2way	x 2way	–	–	x	x
BQ, BDQ	–	–	–	-	–	x	x
BS, BDS US, UDS	x	x	–	x	x	x	x
BT, BDT UT, UDT	x 2way	x 2way	x 2way	–	x	x	–
D1D2, 3	x ----- Relay purposes only - - - - -	x	x	x	x	x	x
(F, G, H, J, K, L, M, N, P, Q, R) TO DA	–	–	–	–	x	x	x

X – means data link is required. Required link availability is nominally continuous coverage = 0.98. This does not include occultation outages which are tbd, depending on specific link geometries. Availability is not utilization.

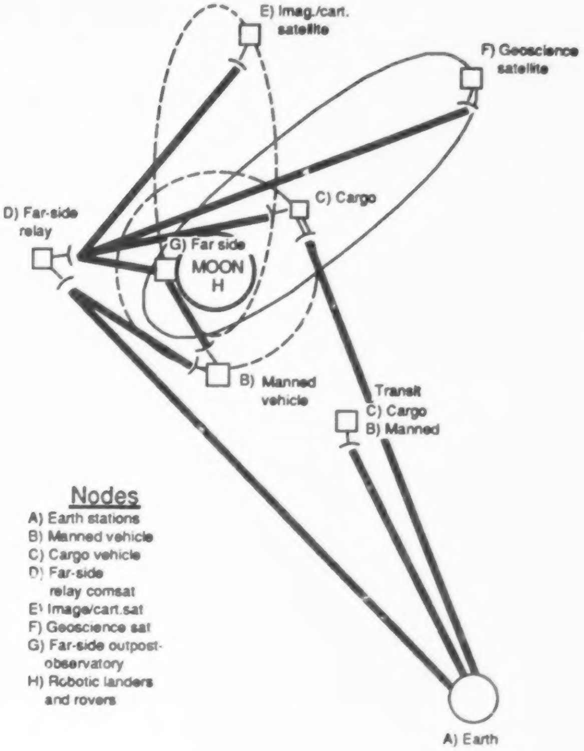
Figure 3.5.2-3.- Lunar Outpost to Early Mars Evolution, Mars portion: comm links.

involving both U.S. and international cooperative spacecraft. Finally, the approximate 6-year extension in development time available in the evolutionary case study will most likely result in more advanced technology in the areas of optical communications, precision Mars navigation, and relay communications satellite onboard data storage and data processing.

3.5.2.3 Lunar-Related Case Studies

Lunar Observatory. The communication architecture for the Lunar Observatory Case Study is shown in figure 3.5.2-4. Like the Mars case studies, it will require voice, video, and data communications. However, due to the short sortie times of the lunar crews, the requirement for continuous communication will be for 10- to 20-day periods only. After crew departure, the activated observatory will require a continuous link with Earth at all times for scientific data transmission and command uplink. Earth-Moon link geometries and gravitational dynamics suggest implementation of a communications relay satellite in halo orbit around the L2 libration point to provide continuous coverage via a single relay from the far side of the Moon to Earth. Careful investigation of alternative L2 orbits is needed to select one that will minimize the delta V needed for stationkeeping. Additional relay spacecraft positions such as L1 and L4 libration points need to be studied. The intent is to arrive at a network and geometry of space and lunar surface relays that will effectively support all near-side and far-side activities. Like the Mars case studies, a communications link will be required between the Earth and the manned vehicle at all times, as well as between the manned vehicle in lunar orbit and the surface outpost. Telemetry and science data communications with the various scientific satellites in lunar orbit, as well as a link to ascertain the health of the cargo vehicle, will be required.

Due to the much shorter communication distance between the Earth and Moon relative to the Earth-Mars distance, a much lower communication frequency and lower power can be employed on the Earth-Moon link. This is subject to frequency allocation and bandwidth constraints imposed by international radiofrequency (RF) spectrum allocation agreements. Combined high-rate video and science data rates of 100 to 300 Mb/s (uncompressed) dominate the Earth-Moon link capacity requirements, but again because of the reduced communications range, this burdens link bandwidth requirements, not effective isotropic radiated power (EIRP) require-



Data type / Link	Video high rate	Video low rate	Voice	Tele-robotics	Sci. data	Eng. term	Cmd, datas load
AB, ADB	x 2 way	x 2 way	x 2 way	-	x	x	x
AC, ADC BC,BDC	-	-	-		x	x	-
EDA	-	-	-		x	x	-
FDA	-	-	-		x	x	-
GDA	x 2 way	-	x 2 way		x	x	-
HDA	x 2 way	-	-	x	x	x	-
BF, BDG	x 2 way	-	x 2 way	-	x	-	-

x Means data link is required. Required link availability is nominally continuous coverage = 0.98. This does not include occultation outages which are tbd, depending on specific link geometries. Availability is not utilization.

Figure 3.5.2-4.- Lunar observatory: comm links.

ments. The lunar observatories on the far side of the Moon require an interference-free RF environment suggesting the use of very high frequencies, such as optical, for the link from the lunar relay satellite to the surface of the far side of the Moon.

Lunar Outpost to Early Mars Evolution (Lunar Portion). The communication architecture for this portion of the evolutionary case study is presented as figure 3.5.2-5. This case study has all of the requirements of the Lunar Observatory Case Study, plus the additional communication requirements associated with the lunar near-side surface operations. It also has the requirement to support communications between the lunar outpost and remote vehicles, crew, and equipment. Finally, additional telemetry and remote control will be required for LLOX mining and resupply ship navigation.

Both lunar case studies will require the tracking of vehicles in lunar orbit, and each case study calls for repeated precision landings (±3 m) at the same site. Navigation and control issues have not been addressed in this analysis to date.

3.5.2.4 Technology Development Needs

Preliminary estimates of telecommunications, navigation, and information management technology development needs are listed in table 3.5.2-II. As would be expected, case 1 needs the least enabling technology. Yet, to meet the expected requirements, a significant improvement in telecommunications capability is required over the current deep-space Voyager link performance. Case Study 4, Lunar Outpost to Early Mars Evolution, requires intensive enabling technology, but more time is available for development.

Specific navigation data type needs are given in table 3.5.2-III.

3.5.3 Prerequisite Program Accommodation of Case Studies

3.5.3.1 Mars-Earth Direct Links

Table 3.5.3-I presents the achievable raw-data rate on a 1-au Mars-Earth telecomm link comparing X-band, Ka-band, and optical communications performance. The X-band capabilities assume the use of existing 70-m and 34-m Deep Space Network (DSN) antenna configurations. The Ka-band performance numbers assume the completed development and implementation of antenna upgrades and Ka-band receivers. The optical frequency capabilities assume a 10-m optical receiving telescope in Earth orbit or on the Earth surface, and a 0.5-m to 1.0-m transmitting telescope at Mars. Additional assumptions

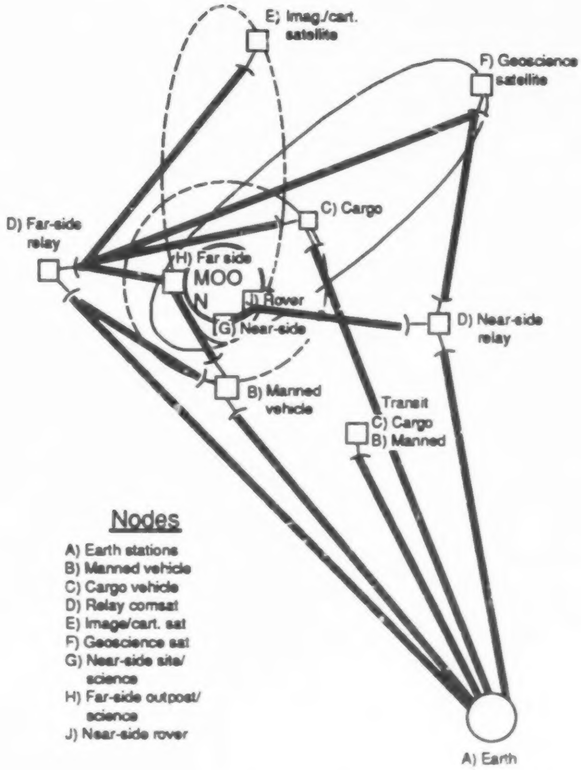

E) Imag./cart. satellite
F) Geoscience satellite
C) Cargo
D) Far-side relay
H) Far side
MOON
J) Rover
G) Near-side
D) Near-side relay
B) Manned vehicle
Transit
C) Cargo
B) Manned
Nodes
A) Earth stations
B) Manned vehicle
C) Cargo vehicle
D) Relay comsat
E) Image/cart. sat
F) Geoscience sat
G) Near-side site/ science
H) Far-side outpost/ science
J) Near-side rover
A) Earth

Table of data types: Connectivity par link

Data type / Link	Video high rate	Video low rate	Voice	Tele-robo-tics	Sci. data	Eng. telm	Cmd, data load
AB, ADB	x 2way	x 2way	x 2way	–	x	x	x
AC,ADC BC,BDC	–	–	–	–	–	x	x
GJ, GDJ	x	x	x 2way	x	x	x	x
EDA	–	–	–	–	x	x	x
FDA	–	–	–	–	x	x	x
GA, GDA	x 2way	x	x 2way	–	x	x	x
HDA	x 2way	–	x 2way	–	x	x	x
JDA	x	–	x 2way	–	x	x	x
BG, BDG BH, BDH	–	–	x 2way	–	x	x	–
GDH	x 2way	x 2way	x 2way	–	x 2way	–	x 2way

x Means data link is required. Required link availability is nominally continuous coverage = 0.98. This does not include occultation outages which are tbd, depending on specific link geometries. Availability is not utilization.

Figure 3.5.2-5.- Lunar Outpost to Early Mars Evolution, lunar portion: comm links

TABLE 3.5.2-II.- TNIM TECHNOLOGY DEVELOPMENT NEEDS (PRELIMINARY)

Need	Case Study			
	1	2	3	4
• Assess technology needs to provide a 20-30 dB increase above Voyager link baseline overall performance for high-rate (>100 Mb/s) Mars-Earth link.	X	X	X	X
• Assess Moon and Mars-Earth/intra system telecom needs and performance, comparing optical, mm-wave, microwave links performance, and needed technology.		X	X	X
• Develop technology needed for the frequencies selected (transmitters, propagation, receivers/ transponders, RF/optical antennas, and pointing.	X	X	X	X
• Analyze weather statistics of Earth-orbital vs Earth-based optical receiver link option.				X
• Assess technology for data compression, coding, and modulation, and data storage needs for all links; initiate development as necessary.	X	X	X	X
• Assess technology needs for precision in situ intra-Mars/ lunar systems orbital, entry, landing, and surface navigation.		X	X	X
• Develop autonomous data link status monitoring and switching on relay orbiters.				X
• Analyze types of orbits for Mars and lunar navigation and relay satellites, including stationkeeping issues.	X	X	X	X

TABLE 3.5.2-III.- NAVIGATION DATA TYPE NEEDS

Data type	Scenario			
	Phobos	Mars	Moon	Evolutionary Moon-Mars
• Doppler	X	X	X	X
• Ranging	X	X	X	X
• VLBI types	X	X		
• Local timing			X	X
• Global positioning				X

are described in the table notes. X-band communications can support most data types except uncompressed high-rate video or high-rate science links that require >6 Mb/s. The addition of Ka-band affords a factor of 5 improvement over X-band performance but will be unable to support multiple high-rate video or science channels greater than 100 Mb/s without 10-20 dB of data compression. Finally, the estimates of optical link performance show an additional factor of at least 10 improvement over Ka-band performance and can support a data rate of greater than 1000 Mb/s. Hence, this link can support a substantial combination of all data types. These performance capabilities indicate that telecommunications upgrades in the next decade should concentrate on development of Ka-band and optical links, and efficient data compression techniques.

3.5.3.2 Intra-Mars Communications Links

As depicted in figures 3.5.3-1 through 3.5.3-3, the proposed Mars missions will require significant link capabilities between equipment and personnel within the Mars system. Example point design links that can service the surface and orbit communication nodes are shown in figure 3.5.3-1; although they do not cover the full range of specific communications requirements, these link designs highlight basic tradeoffs for all intra-Mars communications. As might be expected, there is no single communications frequency or spacecraft antenna coverage pattern that best serves all the intra-Mars communications needs.

Fixed-Surface to Fixed-Surface via Relay Satellite. Fixed or mobile sites on the martian surface may communicate via direct line-of-sight communications only if proximity and local terrain permit. The lack of a significant martian ionosphere for HF ionospheric skip communications necessitates the use of a Mars orbiting satellite to relay communications between non-line-of-sight nodes, fixed or mobile. Figure 3.5.3-1, part A, depicts a Ka-band link between two fixed sites on the martian surface through such a relay satellite at a Mars-synchronous altitude of

TABLE 3.5.3-I.- MARS-EARTH DIRECT LINKS RAW DATA-RATE CAPABILITIES

Receiver system Transmitter	Baseline X-band 70 m/34 m	Extended * present tech Ka-band ** 70 m/34 m	New tech potential Optical ** 10-m telescope
1 S/C at Mars (note 1.1)	6/1.5 Mb/s (note 1.2)	100/25 Mb/s (note 1.3)	1000 Mb/s
2 Mars surface fixed site	29/7.2 Mb/s (note 2.1)	100/25 Mb/s (note 2.2)	1000 Mb/s (note 2.3)
3 Mars surface rover	0.12/0.029 Mb/s (note 3.1)	0.4/0.1 Mb/s (note 3.2)	

Notes:

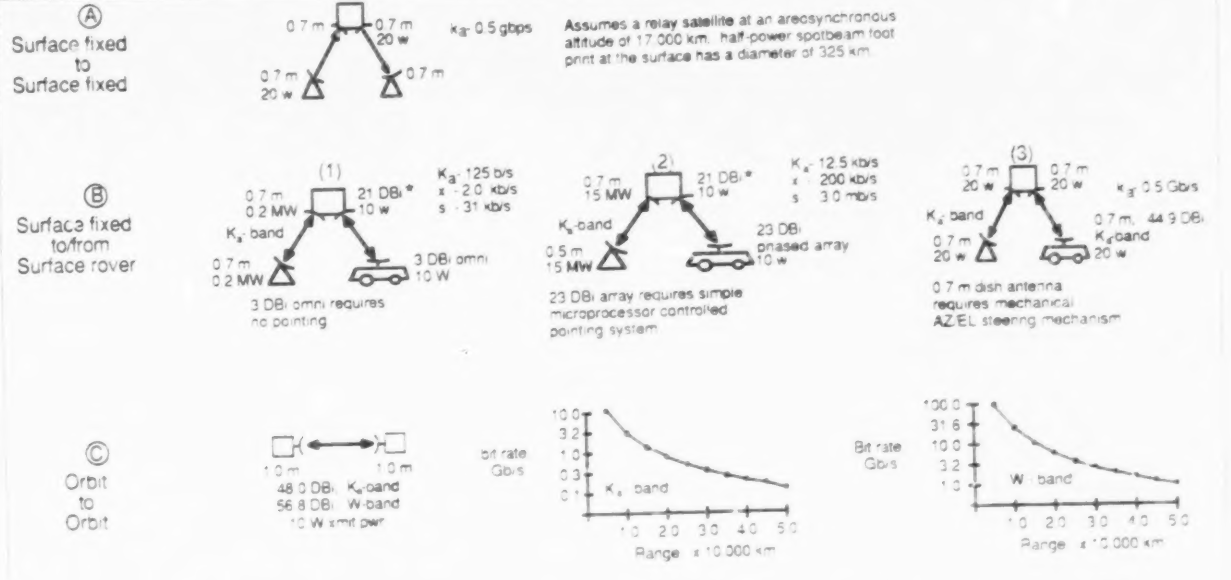
1.1 S/C: 3.6-m ant, 8.4 GHz, 20 W, 1.0 au; Voyager class S/C
1.2 S/C: 5.0-m ant, 32.4 GHz, 50 W, 1.0 au
1.3 S/C: 0.5-m telescope, 0.532 micron, 12 W, 1.0 au reduced S/C weight, power, and volume can be achieved by trading off link performance

2.1 Mars surface fixed: 5.0-m ant, 8.4 GHz, 50 W, 1.0 au
2.2 Mars surface fixed: 5.0-m ant, 32.4 GHz, 50 W, 1.0 au
2.3 Mars surface fixed: 1.0-m telescope, 0.532 micron, 6 W, 1.0 au

3.1 Mars surface rover: 0.5-m ant, 8.4 GHz, 20 W, 1.0 au
3.2 Mars surface rover: 0.5-m ant, 32.4 GHz, 20 W, 1.0 au

* Uplink capability is over 1 Gb/s at 8.4 GHz, 70-m ant, 100 kW, but frequency allocation and modulation bandwidth limited

** 10-15 dB of data compression is possible on all links as proven algorithms become available.


Ⓐ
Surface fixed to Surface fixed
0.7 m
0.7 m 20 w
K_a- 0.5 gbps
0.7 m 20 w
0.7 m
Assumes a relay satellite at an areosynchronous altitude of 17,000 km. half-power spotbeam foot print at the surface has a diameter of 325 km
Ⓑ
Surface fixed to/from Surface rover
(1)
0.7 m 0.2 MW
21 DBi* 10 w
K_a- 125 b/s
x - 2.0 kb/s
s - 31 kb/s
K_a- band
0.7 m 0.2 MW
3 DBi omni 10 W
3 DBi omni requires no pointing
(2)
0.7 m 15 MW
21 DBi* 10 w
K_a- 12.5 kb/s
x - 200 kb/s
s - 3.0 mb/s
K_a-band
0.5 m 15 MW
23 DBi phased array 10 w
23 DBi array requires simple microprocessor controlled pointing system
(3)
0.7 m 20 w
0.7 m 20 w
K_a- 0.5 Gb/s
K_a- band
0.7 m 20 w
0.7 m, 44.9 DBi K_a-band 20 w
0.7 m dish antenna requires mechanical AZ/EL steering mechanism
Ⓒ
Orbit to Orbit
1.0 m
1.0 m
48.0 DBi K_a-band
56.8 DBi W-band
10 W xmit pwr
bit rate Gb/s
10.0
3.2
1.0
0.3
0.1
K_a band
1.0
2.0
3.0
4.0
5.0
Range x 10,000 km
Bit rate Gb/s
100.0
31.6
10.0
3.2
1.0
W band
1.0
2.0
3.0
4.0
5.0
Range x 10,000 km

Notes
1. W-band = 90 ghz. K_a-band = 32.4 GHz. X-band = 8.4 GHz. S-band = 2.1 GHz
2. Receive system noise temperature = 300 K for all cases
3. All relay satellites are assumed to be at a slant range of 18,000 km from surface nodes

*21 DBi s/c antenna provides coverage of the entire martian surface visible from areosynchronus orbit

Figure 3.5.3-1.- Intra-Mars comm links raw data rate capabilities.

17,000 km. The Ka-band frequency is chosen to allow a high-rate 0.5-Gb/s communications capability. This results in a narrow beam that covers a 300-km-diameter circular area on the Mars surface. Higher gain antennas or higher communications frequencies may be employed to increase link capacity further, again at the expense of decreasing beam footprint size. If the two communications nodes are not within the relay antenna beam, an additional beam will be required. The alternative is to use a spacecraft antenna that provides broad hemispherical coverage of the martian surface and hence results in a 20-dB to 30-dB decrease in link capability.

Surface-Fixed to/from Surface Rover via Relay Spacecraft. There are several important tradeoffs for rover support communications. A surface rover can use either an actively pointed antenna or an omnidirectional antenna to maintain communications to an in-orbit relay spacecraft as the rover changes azimuth orientation and traverses rough terrain. Figure 3.5.3-1, part B (1), shows that lower communications frequencies achieve the best link performance in the satellite relay employs a fixed-gain antenna covering the visible martian hemisphere and the rover employs an omnidirectional antenna. The telecomm capacity of this arrangement, however, is insufficient to support a telerobotics video link. This type of link would be valuable for collection of scientific data from Mars surface science instruments for relay to a central data collection site. The same figure, part B (2), depicts a similar arrangement with the rover now employing a 23-dBi-gain steerable array. The link performance of this arrangement is improved but still would require 15-20 dB of video data compression to be a workable telerobotics link. Finally, figure 3.5.3-1, part B (3), shows a 0.5 Gb/s link that could support the telerobotics link. The penalty is the need to maintain accurate antenna pointing with a narrow (less than 1 degree) beamwidth antenna while the rover is in motion. Higher frequencies may be employed on the rover-space link to reduce the size of the rover antenna. In addition, the narrower beamwidth of the higher gain spacecraft antenna will require a steerable beam. These diagrams highlight some of the technical difficulties and tradeoffs that will require more detailed study.

Round-trip light time for a surface-to-relay-to-rover link is roughly 0.1 s, perhaps compatible with remote piloting needs of the rover.

Fixed Surface or Rover to/from Orbit. A single leg of any of the relay links depicted in figure 3.5.3-1 may be considered a good approximation of orbit-surface link design needs.

Orbit-to-Orbit. An orbit-to-orbit link using a second relay satellite would be employed to relay data and control information from one side of the Mars surface/orbit to the other or for additional relay back to Earth.

Figure 3.5.3-1, part C, shows link capacities as a function of interspacecraft range for both Ka-band and W-band communications frequencies. It may be desirable to use optical communications frequencies for these links to reduce spacecraft power and weight requirements.

Moon-Earth Direct Links. Table 3.5.3-II shows example link capabilities for Moon-to-Earth direct links. Because of the greatly reduced range compared to the Mars-Earth range, adequate link capacity is available with existing communications technology. Selection of the appropriate link configuration will be based on issues such as spacecraft weight and power constraints, international frequency allocation agreements, and bandwidth and technology availability constraints.

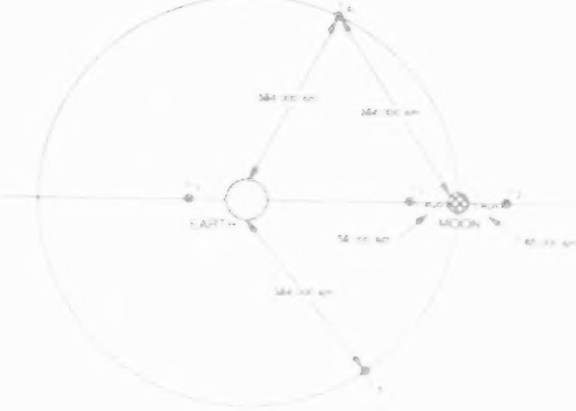


Figure 3.5.3-2. Location of the 5 libration points in the Earth-Moon System

Intralunar Communications Links. The Sun-Earth-Moon gravitational interactions do not allow stationary orbits around the Moon similar to those possible around the Earth or Mars. Figure 3.5.3-2 depicts the five libration points in the Earth-Moon system where the net gravitational pull and the centrifugal force just balance. Of these five points only L4 and L5 are truly stable. A relay satellite at L4 or L5, if slightly disturbed, would oscillate around its original position. The other three points are quasi-stable in that a slight disturbance would cause the relay satellite to drift away from the libration point. L1 is a logical choice for a lunar near-side relay satellite if needed. Ranges to the Moon and Earth are 58,000 km and 226,000 km, respectively. A relay satellite in halo orbit around L2 would provide a relay link with visibility to the entire far side of the Moon and visibility to the Earth and the near-side relay satellite at L1. Ranges to the Moon and Earth for an L2 relay are 65,000 km and 449,000 km, respectively. The stationkeeping delta-V requirements for such a spacecraft are a topic of further investigation. Relay satellites at L4 or L5 would have greatly reduced stationkeeping delta-V requirements but would afford only a meager view of the lunar far side. Ranges to the

TABLE 3.5.3-II.- LUNAR-EARTH DIRECT LINKS RAW DATA-RATE CAPABILITIES

Receiver system / Transmitter system	Required data rate (uncompressed)	Baseline X-band 34 m	Extended present tech Ka-band 34 m	New tech potential Optical 1-m telescope
1 S/C at Moon	150 Mb/s (note 1.1)	150 Mb/s (note 1.2)	150 Mb/s (note 1.3)	1000 Mb/s
2 Moon surface	400 Mb/s fixed site	400 Mb/s (note 2.1)	400 Mb/s (note 2.2)	1000 Mb/s (note 2.3)
3 Moon surface	150 Mb/s rover (note 3.1)	150 Mb/s (note 3.2)	150 Mb/s	

Notes:

1.1 S/C: 1.0-m ant, 8.4 GHz, 2.5 W
1.2 S/C: 1.0-m ant, 32.4 GHz, 0.5 W
1.3 S/C: 0.20-m telescope, 0.532 micron, 0.5 W

2.1 Lunar surface fixed: 1.0-m ant, 8.4 GHz, 50 W, 1.0 au
2.2 Lunar surface fixed: 1.0-m ant, 32.4 GHz, 50 W, 1.0 au
2.3 Lunar surface fixed: 0.20-m telescope, 0.532 micron, 1.3 W

3.1 Moon surface rover: 0.5-m ant, 8.4 GHz, 10 W
3.2 Moon surface rover: 0.5-m ant, 32.4 GHz, 2 W

Range = 400,000 km

* Higher rate capability may be used to reduce space vehicle weight, power, and volume.

Moon and Earth for an L4/L5 relay are 384,000 km and 384,000 km, respectively.

Figure 3.5.3-3 depicts typical intralunar communications links assuming relay satellites at the L1 and L2 (halo orbit) libration points.

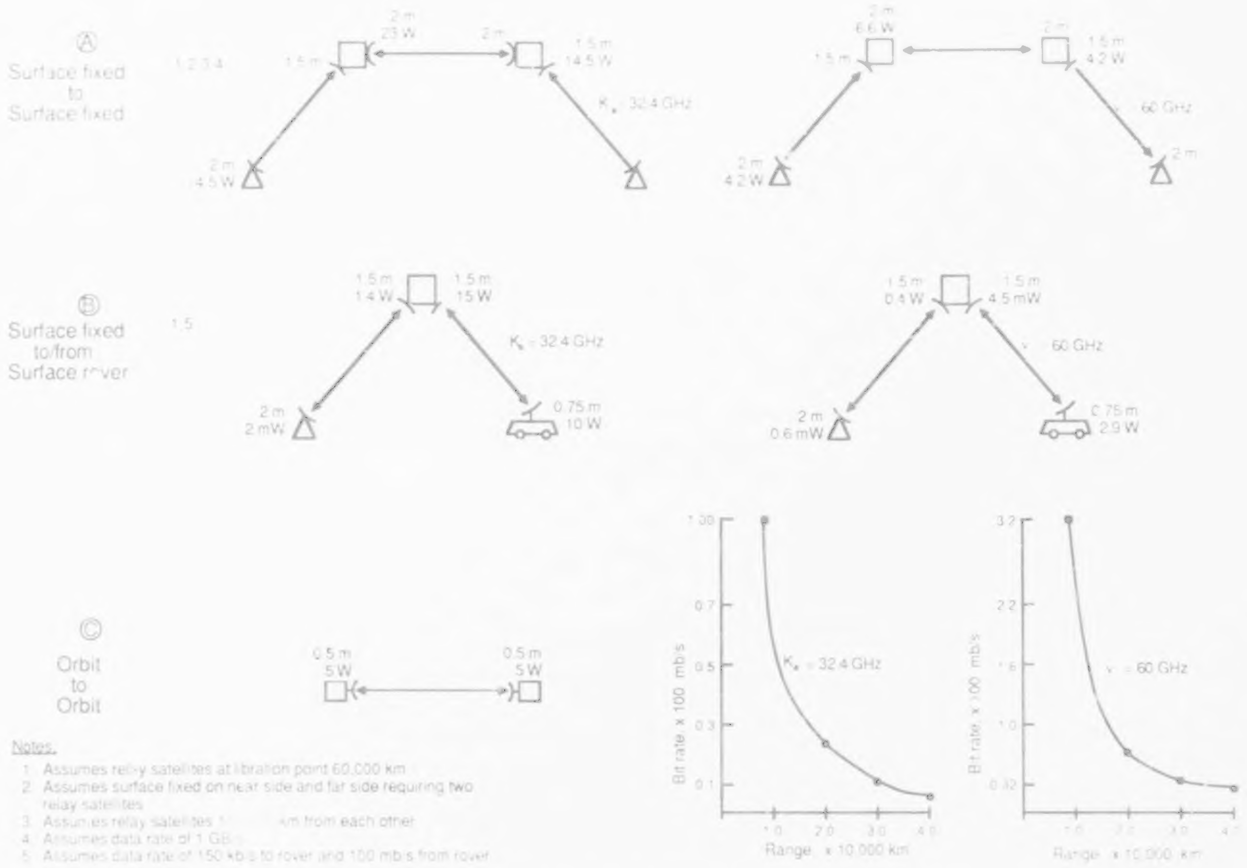
Surface-Fixed to Surface-Fixed via Relay Satellite. Figure 3.5.3-3, part A, shows Ka-band and W-band alternatives for achieving 1.0 Gb/s capacity from the near to the far side of the Moon. Use of W-band reduces spacecraft power and weight requirements. Use of optical frequencies would further reduce spacecraft power and weight requirements and avoid RF interference to sensitive radio telescope equipment on the lunar far side. The high-gain spacecraft antennas proposed in these model links have a half-power beamwidth of 0.43 and 0.23 degrees for Ka-band and W-band, respectively, corresponding to lunar surface beam footprints of diameter 450 km and 240 km, respectively. Hence multiple spotbeams generated by a phased array may be required to cover diversely located surface sites. Surface-to-surface lunar communications within the same hemisphere would employ only a single relay satellite and require essentially the same antenna size and transmit power combinations shown for the dual relay link of figure 3.5.3-3, part A.

Surface-Fixed to/from Surface Rover via Relay Satellite. Figure 3.5.3-3, part B, depicts Ka-band and W-band alternatives for achieving 0.5 Gb/s capacity between a rover and a fixed lunar surface site in the same hemisphere. Again the narrow beamwidth satellite antennas cover only a small lunar surface area, thus requiring a multiple beam spacecraft antenna and/or spacecraft antenna steering to accommodate multiple rovers and wide area rover movement. Rover antennas are also narrow beamwidth, which complicates pointing as the rover traverses rough terrain.

Round-trip light time for a lunar surface-L2 relay-rover link is roughly 0.8 s and may preclude remote piloting options.

Surface to/from Orbit. Data links are to be determined by spacecraft orbit and communication requirements.

Orbit-to-Orbit. Figure 3.5.3-3, part C, depicts a generic spacecraft-to-spacecraft link. Accompanying graphs



Ⓐ
Surface fixed to Surface fixed
1 2 3 4
2 m
23 W
2 m
1.5 m
1.5 m
14.5 W
K_a = 32.4 GHz
2 m
4.5 W
2 m
6.6 W
2 m
1.5 m
1.5 m
4.2 W
60 GHz
2 m
4.2 W
2 m
Ⓑ
Surface fixed to/from Surface rover
1 5
1.5 m
1.4 W
1.5 m
15 W
K_a = 32.4 GHz
2 m
2 mW
0.75 m
10 W
1.5 m
0.4 W
1.5 m
4.5 mW
60 GHz
2 m
0.6 mW
0.75 m
2.9 W
Ⓒ
Orbit to Orbit
0.5 m
5 W
0.5 m
5 W
1.00
0.7
0.5
0.3
0.1
Bit rate, x 100 mb/s
K_a = 32.4 GHz
1.0
2.0
3.0
4.0
Range x 10,000 km
3.2
2.2
1.6
1.0
0.32
Bit rate, x 100 mb/s
= 60 GHz
1.0
2.0
3.0
4.0
Range x 10,000 km

Notes:

1. Assumes relay satellites at libration point 60,000 km
2. Assumes surface fixed on near side and far side requiring two relay satellites
3. Assumes relay satellites [illegible] km from each other
4. Assumes data rate of 1 GB
5. Assumes data rate of 150 kb/s to rover and 100 mb/s from rover

Figure 3.5.3-3 - Intralunar comm links raw data rate capabilities.

display telecomm capacity as a function of range for Ka-band and W-band. Other details depend on spacecraft orbits and communication requirements.

3.5.4 Support Required from Other NASA Organizations

OSO requires support from the Office of Aeronautics and Space Technology (OAST) in the following areas:

a. Spacecraft K$_a$-band and optical communications technology development and advanced engineering in:
    (1) Antennas and electronic beam steering
    (2) High-power K$_a$-band and optical transmitters
    (3) Microwave/optical transponders
b. Onboard memory to reduce peak-to-average channel loading
c. Efficient data compression techniques
d. Telerobotics information management techniques
e. Demonstration opportunities for all new technology prior to phase C/D specification

Further definition of OEXP needs in the following areas should also be provided to OSO.

a. Data rate and quality requirements for mission science
b. Planetary lunar delivery and in situ navigation accuracy requirements for both real- and non-real-time functions.
c. Mission operations data requirements including video data rates
d. Operations strategies
e. Location of operations centers
f. Mission sequences
g. Related information-processing requirements

3.5.5 Conclusions

3.5.5.1 Systems Issues

Key systems issues concern the allocation of in situ telecommunications and navigation responsibilities to OSO or to the manned mission. If there is significant remote-location activity on a large number of missions, it may be appropriate to have OSO provide the services; for example, a Mars-centered TNIM system could be available to all users. Examples of Earth-centered systems providing such services are the DSN and the Tracking and Data Relay System (TDRS). Further discussion on this subject is found in tables 3.5.5-I and -II.

3.5.5.2 Organizational Issues

OSO suggests the establishment of two new OEXP study agents: an Integration Agent (IA) for Mission Operations and TNIM Special Assessment Agent (SAA). The proposed lead center(s) and tasks are listed:

a. A Missions Operations IA may be required in FY 1988 and beyond. Suggested centers are JSC with JPL/GSFC support for TNIM. Tasks are to
    (1) Determine locations, requirements, and designs of "centers of operations" per case study phase.
    (2) Develop operations plans and approaches for each case study including mission sequences, logistics, resource scheduling, and assembly operations and requirements.
    (3) Develop requirements concepts and options for up/downlink TNIM including human/human and human/machine interfaces.
    (4) Expand the working interface between the OSO system designer and OEXP mission and system designers to include assignment of appropriate system responsibilities.
b. A TNIM SAA may be needed in FY 1989 and beyond. Suggested centers are JPL/GSFC. Tasks are to
    (1) Assess TNIM system design.
    (2) Recommend TNIM system design trade studies, standards, and options.
    (3) Assess TNIM technologies and their potential impact to case studies development.

3.5.5.3 Technical Issues

A summary of current technical issues which have been raised in the course of OSO activities are listed:

a. Methodologies for merging science, operations, and engineering data into a single data stream
b. Requirements for continuous telecomm and nav per channel; this drives relay needs
c. Requirements for reliability and functional redundancy
d. Onboard TNIM requirements: weight power, volume, spacecraft stability, interfaces
e. Data compression potential for various data types
f. Local vs. global navigation requirements
g. Mission-dedicated vs. NASA institutional operations support systems
h. Flight mechanics issues
    (1) Impact on navigation systems of planetary landing requirements. Options for meeting navigation requirements include local beacons or GPS systems. Case Studies 2, 3, and 4 are affected.
    (2) The method for providing lunar relay communications. Satellite location/orbit, orbit maintenance delta V, inherent reliability, and orbit stability need to be examined.

TABLE 3.5.5-I.- DEEP-SPACE TELECOM RELAY ARCHITECTURAL DESIGN

High-level architectural design principles for a deep-space telecom relay

- Provides multimission support using Mars and Moon local relay and navigation instrumentation, and covers
  - Manned missions
  - Unmanned science missions
  - Approved international cooperative missions
- Incorporates long-term investment concepts in facilities on Earth and in space; system engineering based on life-cycle costs
- Offers an expected 10- to 30-year lifetime of facilities on station
- Provides evolutionary performance upgrades when needed, but no capability is discarded or unused
- Provides redundancy through system design; basic principle is that telecom and navigation are robust and fail last; risk is managed in the system design
- Offers design, implementation, operation, and upgrade by OSO
- Provides transportation by appropriate NASA program/missions

TABLE 3.5.5-II.- A DEEP-SPACE TELECOM RELAY CONCEPT

- Makes service available to all users in Mars/Moon vicinity
- Merges/distributes traffic from/to many local terminals in remote system to/from Earth which permits use of much lower cost telecommunications and navigation capabilities in supported vehicle/landers
- Provides communication paths free from occultation for surface and orbiting terminals
- Supports cross-links to other relays for failsoft backup
- Controls operation of OSO local navigation network/orbiter functions; incorporates appropriate intrasystem navigation functions
- Incorporates autonomous monitoring, merging, storing, and routing functions; also provides some data processing capabilities for users
- Uses frequencies appropriate to local and Earth link needs, but can include a "bent pipe" mode

3.6 EXPLORATION IMPACTS TO SPACE STATION FREEDOM EVOLUTION

3.6.1 Role of the Office of Space Station (OSS) in Manned Exploration

3.6.1.1 Policy

The OSS (code S) has the responsibility to define the evolution of Space Station Freedom facilities necessary to accommodate the four Agency exploration case studies under consideration. This responsibility follows from the Presidential directive on National Space Policy of February 11, 1988, which states that the "Space station will allow evolution in keeping with the needs of station users and the long-term goals of the United States."

Using prerequisites for a low-Earth-orbit (LEO) transportation node provided by the Office of Exploration (OEXP), OSS has assessed the impacts to the Space Station Freedom program of supporting the individual scenarios. Detailed assessments of the implications of supporting each scenario and implementation plans required to accommodate the scenarios are discussed in subsequent sections to the depth feasible at this time. In general terms, preliminary approaches for implementing the exploration requirements include derivation of infrastructure concepts as well as a levy of formal growth requirements on the baseline station. Additionally, a small-scale advanced development program has been inaugurated. Equally as important, the long-range planning mechanisms are in place to ensure that the space station has the ability to co-evolve with the space infrastructure necessary to support manned exploration outside Earth orbit.

3.6.1.2 Responsibilities

Activities for Space Station Freedom evolution planning and advanced development are collected under a separate task known as Transition Definition. There are two elements to this program: (1) system studies and (2) analysis and advanced development.

Responsibility for both technical planning and fiscal management of the Transition Definition program reside in the NASA Headquarters Strategic Plans and Programs Division (SPPD) (level 1), apart from the level 2 Space Station Office at Reston, Virginia. This placement of responsibility allows for visibility of evolution policy and a thorough and independent analysis of the requirements posed on station systems by long-range missions. The SPPD is responsible for the maintenance of the document, "Space Station Evolution—A Technical and Management Plan." Level 2 retains cognizance over the actual management of design and development of baseline station evolution provisions.

The Evolution Definition Office (EDO) at the Langley Research Center (LaRC) is the engineering arm of level 1 station for pre-phase-A activities related to system studies and analysis. The head of the EDO chairs the NASA-wide Evolution Working Group (EWG) which provides intra-Agency communication and coordination for evolution planning. A key requirement for this group is to interface with the baseline work-package (level 3) regarding evolution matters.

3.6.1.3 Program Objectives

The overarching goal of the Transition Definition element of the Space Station Freedom program is to pursue those activities necessary to define and prepare for station evolution in keeping with the needs of users and long-term national goals. The overall objectives of the Transition Definition program are

a. To define station evolution configurations consistent with user requirements and program constraints

b. To define and incorporate baseline design accommodations (hooks and scars) to satisfy evolution requirements

c. To develop advanced technology that ensures technology readiness to enhance station capabilities and to enable station evolution

3.6.1.4 Program Strategy

The strategy chosen to implement the objectives relies on understanding future space options and the implications of these on today's decisions. The challenge to plan for Space Station Freedom evolution is to understand the probable evolution paths and the corresponding infrastructure options to the extent that current resources can be wisely allocated to the necessary "hooks and scars" and to the appropriate advanced development efforts. Thorough understanding of the forces and constraints requires close coupling of evolution mission requirements, space and ground infrastructure planning, technology development, and external policy imperatives. Reference evolution configurations, which have been subjected to systems/operations analysis, are iterated with the exploration community so that the ramifications of evolution options are understood among all planning groups.

The challenge to provide for Space Station Freedom evolution takes the form of keeping the options open to support future missions. Therefore, planning for evolution is necessarily conducted in parallel with the design and development of the baseline station. The Transition Definition program is facilitating the changes by ensuring that the necessary hooks, scars, and technology transparency are incorporated into the baseline design to allow for various forms of evolution. Further, advanced development activities will be focused according to technology needs identified by evolution studies and analysis. Current emphasis is on maturing applications with a high payoff in enhanced efficiency and productivity. This includes technology developments in advanced automation and telerobotics. Separate working groups exist in systems autonomy and telerobotics to aid the SPPD in planning and executing the advanced development program.

3.6.2 Case Study Needs Assessment

3.6.2.1 Case Study 1

No node is required for Case Study 1, Human Expeditions to Phobos, by OEXP ground rules.

3.6.2.2 Case Study 2

The transportation node in LEO required for Case Study 2, Human Expeditions to Mars, can be accomplished in a number of ways. The five options listed in table 3.6.2-I are currently being studied by the Office of Space Station. They range from full accommodation at station (option 1) to a completely branched facility (option 4) which has no

TABLE 3.6.2-I.- VEHICLE ACCOMMODATION OPTIONS

Option 1:	Station based	- All vehicle accommodations based on station
Option 2:	Station based w/PTF	- Vehicle assembly and refurbishment facility is on-station. Propellant is located on a coorbiting propellant transfer facility
Option 3:	Transportaion depot (Man-tended)	- Vehicle accommodations and propellant are kept on a coorbiting platform, but crew is based on station
Option 4 :	Transportation depot (Permanently manned)	- A separate facility is provided for vehicle, crew, and propellant
Option 5:	Transportation depot w/PTF	- Vehicle assembly accommodations and crew facilities are provided off-station. In addition, a coorbiting propellant transfer facility is used to store and reliquefy propellant onorbit

interaction with the baseline station. Intermediate options include assembly at the station with propellant storage and handling at a coorbiting propellant tank farm (PTF) (option 2) and use of a coorbiting platform as the assembly and propellant storage base with the crew quarters at the station (option 3). The latter requires a space-based and man-rated orbital maneuvering vehicle (OMV) with crew cab.

Before defining Space Station Freedom accommodation of the Mars mission, the following ground rules and assumptions were established:

a. Phase 1 station configuration is used as the baseline.

b. Life science research will be conducted on the station.

c. A heavy lift launch vehicle (HLLV) with a 91-t payload-lift capability is assumed to be available.

d. A space transfer vehicle capable of delivering/maneuvering 91 t in LEO is required.

e. Mars mission vehicles are assembled in LEO; only one vehicle stack is assembled and verified in LEO at a time.

f. Two man-rated OMV's are available for routine crew transfer if remote assembly facilities are necessary (vehicle accommodation option 3).

g. Liquid oxygen and hydrogen are the propellants for the Mars vehicles, hydrazine for the OMV's (with cold gas jets for station proximity operations).

In the process of establishing station capabilities and concepts for accommodation of the Mars mission vehicles, a list of required design features was established.

a. Size and volume to accommodate Mars vehicles and support equipment

b. Pressurized "command center" for controlling and monitoring EVA and robotic activities

c. Capability for expansion

d. Robotic and EVA access to vehicle and propellant tanks

e. Simple vehicle egress/separation
    (1) Vehicle egress along velocity vector or negative radius vector
    (2) Room to avoid collisions with structure

f. Micrometeoroid/impact protection for vehicle, EVA crew, and propellant by maximum possible exclosure of vehicle

g. Containment of debris produced by vehicle processing operations within an enclosure

h. Thermal protection for EVA crew and propeliant
    (1) Enclosed volume if possible
    (2) Shielded propellant tanks

i. Design features specific to depot (accommodation option 4).
    (1) Docking facilities to accommodate OMV and shuttle
    (2) Room for propellant tanks and support equipment
    (3) Solar dynamic power system
    (4) Guidance, navigation, and control (GN&C), communications and tracking (C&T), and reaction control systems (RCS)

In addition to the design features listed above, the following performance goals were established for all concepts.

a. Controllability of all phases of vehicle assembly:
    (1) Minimum control system sizing and complex
    (2) Minimum torque equilibrium angles to maintain protection envelope, ease of separation, and viewing angles

b. Orbit decay of station/depot and vehicle:
    (1) Decay rates of complete vehicle and station/depot to allow safe separation
    (2) Minimum reboost propellant needs for all phases of vehicle assembly

The program milestones for this case study are given in figure 3.6.2-1. The 14-month round-trip mission for piloted vehicles was established in an attempt to reduce crew time under weightless conditions to a duration for which countermeasures to the deleterious effects could be developed. The Space Station Freedom life sciences program to develop these countermeasures is shown to take 5 or more years, starting at permanently manned capability (PMC). The first 2 years are devoted to increasing crew staytime at station to 6 months (a baseline station goal). The next 2 years extend the countermeasures program to 14 months, followed by a 14-month mission simulation. At this time the zero-g countermeasures program would be considered acceptable for the Mars mission. Note that the Mars vehicles are in phase C/D during the period of countermeasure development. If countermeasures cannot be developed, the Mars vehicles would have to be converted to an artificial-g capability or the mission postponed or canceled.

Assembly and checkout of the Mars vehicles will require major advances in automation and telerobotics to reduce the crew requirements to acceptable levels. Studies are currently underway to establish the specific technology advances required and the station resources to accomplish the assembly/checkout function. At this time we can only provide a "best guess" at these requirements.

Table 3.6.2-II presents the station resource requirements for the Mars mission. Here six crew members are assigned to the assembly tasks and operate in two three-person shifts (table 3.6.2.-III). This assumes one crewmember in the assembly/service lab control center and two performing EVA or telerobotic tasks. Accommoda-

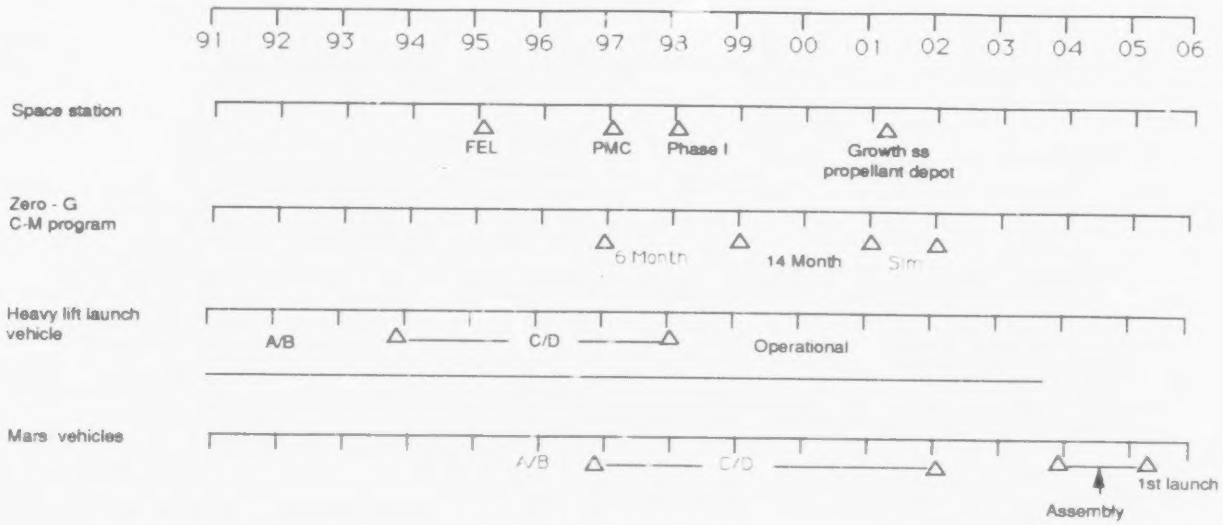


Figure 3.6.2-1.- Program milestones for Space Station Freedom accommodation of Case Study 2.

TABLE 3.6.2-II.- SPACE STATION FREEDOM RESOURCE REQUIREMENTS FOR HUMAN EXPEDITIONS TO MARS

Crew		Facilities	
User: 14	Total: 18	3 U.S. labs:	1 Assembly/serv. lab 1 Human life science 1 CELSS
Power (kw)			
User	Total	1 ESA lab 1 JEM 3 Habs	2 Attached press. payloads 1 Assembly hangar 1 Servicing bay (+ OMV) 1 Quarantine facility
110-130	205-225		

TABLE 3.6.2-III.- AVERAGE SPACE STATION FREEDOM CREW REQUIREMENTS

	'97	'98	'99	'00	'01	'02	'03	'04	'05-'09
Baseline user crew[1]	6	6/6	6/6	6/6	6/6	6/6	6/6	6/6	6/6
Additional life science crew[2]	1.5	1.5/1.5	1.5/2.5	2.5/2.5	2.5/2.5	0.5/0.5	0.5/0.5	0.5/-	-/-
Onorbit technical & development crew	-	2/2	2/2	2/2	2/2	1/1	1/1	1/-	-/-
Vehicle assembly & checkout crew	-	-/-	-/-	-/-	-/3 *	6/6	6/6	6/6	6/6
Total user crew	7.5	9.5		10.5		13.5		12	
Housekeeping crew	2	2		3		4		4	
Total crew onorbit	9.5	11.5		13.5		17.5		16	

[1] Reserved for coexisting science, commercial, and technology activities

[2] 1.5 crew for life science missions embedded in baseline user crew

* Station growth hardware (hangers, SD power, habs & labs, (etc), assy. crew

tion of the basic life sciences program plus the zero-g countermeasures program requires a full lab devoted to human life sciences plus a 4-meter centrifuge located in an attached pressurized facility; later a closed ecological life support system (CELSS) is added to the program. The power and crew requirements shown in table 3.6.2-II also include the requirements for continued operation of the European Space Agency (ESA) and Japanese lab modules. The quarantine facility is an option to accommodate, onorbit, the analysis of samples returned from Mars by an unmanned Mars Rover Sample Return (MRSR) precursor mission (if required).

The servicing bay and OMV are required because the OMV must remotely rendezvous and dock with HLLV payloads and bring them to the station.

Accommodating the life sciences, technology development and assembly/checkout functions require doubling station mass (for accommodation option 2 of table 3.6.2-I) as indicated in table 3.6.2-IV for two concepts. In addition, the Mars vehicles themselves (without propellant) are both large and massive, bringing the total mass at the station to a level of 900 t.

The mass summary for a third concept, based on accommodation option 3, is given in table 3.6.2-V. The open-box configuration appears to be somewhat less massive than the additions to the baseline station given in table 3.6.2-IV. It must be noted, however, that the additional life sciences and technology development activities as well as the two additional hab modules and associated nodes must still be added to the baseline station. The servicing facility and OMV's are also required at station to transport crew and the HLLV payloads. Thus the total mass in space to support the Mars mission is substantially greater (approximately 20 percent) for the accommodation option 3 concept (open box). The configurations are shown in figures 3.6.2-2 through -4.

TABLE 3.6.2-IV.- SPACE STATION FREEDOM GROWTH HARDWARE TO ACCOMMODATE HUMAN EXPEDITIONS TO MARS

Component	Concept A		Concept B	
	Num.	Mass (kg.)	Num.	Mass (kg.)
Truss bays	101	7,400	106	7,800
Utility bays	101	22,700	106	23,600
Solar dynamics	6	33,100	6	33,100
Habitat module	2	39,000	2	39,000
Node	4	23,300	6	34,000
Laboratory module	2	62,900	2	62,900
Pocket labs	2	10,200	2	10,200
Servicing facility + 2 MOS (wet)	1	22,400	1	22,400
Standard airlock	1	2,000	1	2,000
Hangar + equipment	1	25,000	1	25,000
Total additional hardware		248,000		260,000
+ Phase 1 station 210,000 kg.		458,000		470,000
+ Mars cargo vehicle 231,000 kg.	1	689,000	1	701,000
OR				
+ Mars piloted vehicle 439,000 kg.	1	879,000	1	909,100

TABLE 3.6.2-V.- BRANCHED TRANSPORTATION NODE MASS SUMMARY — OPEN BOX CONFIGURATION

Component Name	Mass (kg)
Airlock	2,014
Alpha joints	1,200
CMG's	1,567
Cupola	1,455
Docking adapters	1,000
Nodes (2)	9,091
Command center	31,523
MSC/transporter	4,909
RCS clusters	1,025
RCS propellant & tanks	6,364
Solar dynamic power modules (2)	14,078
TDRSS & antenna	586
Teleoperated Servicer (2)	2,381
Propellant storage tanks (11)	68,924
Attached hardware	12,980
Radiators (2)	3,670
Logistics	8,285
Truss	9,875
Utility trays	18,008
Total	198,900

From the station viewpoint, vehicle controllability and onorbit lifetime are first-order concerns in the test for concept feasibility. An orbiting spacecraft, such as the station or the transportation depot, is subject to a variety of environmental effects which disturb its flight attitude. Such disturbances include forces due to aerodynamic drag, the difference in gravitational force due to the mass distribution (gravity-gradient forces), and forces due to solar radiation pressure. The net result of these disturbances is a buildup of angular momentum which must somehow be countered to maintain the desired flight attitude.

The scheme for managing momentum buildup in the current Space Station Freedom design involves the use of control moment gyros (CMG's) as a method of storing momentum. As the station passes through each orbit, a

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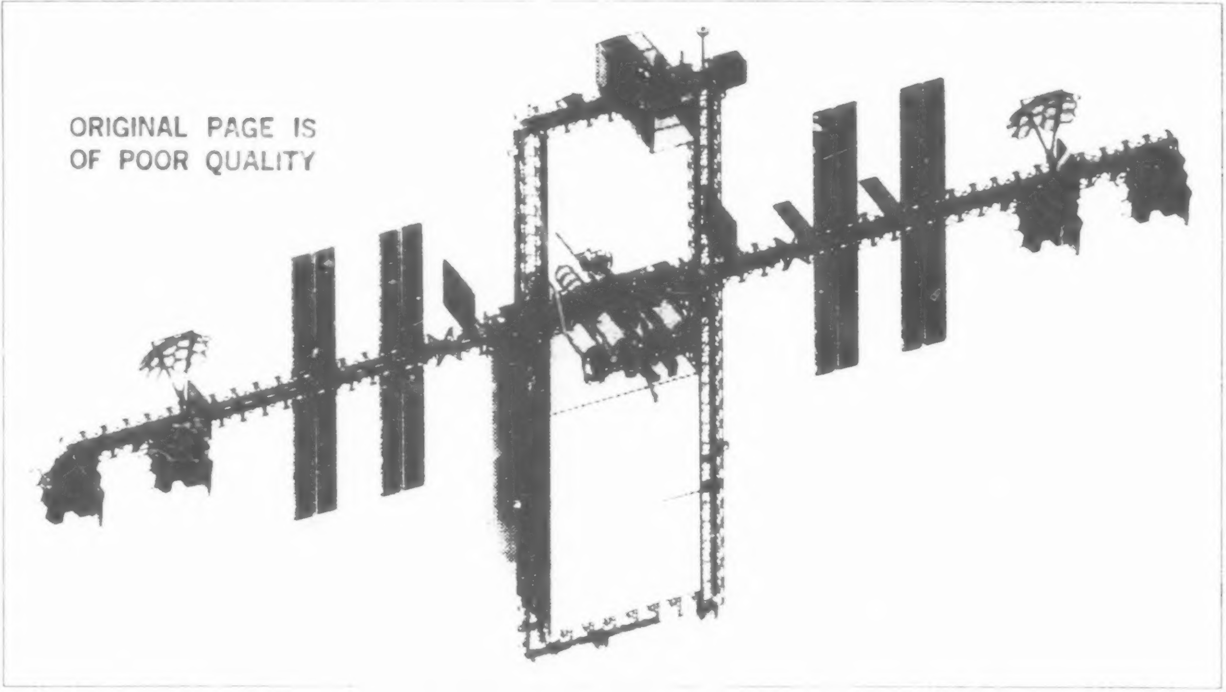

Figure 3.6.2-2.- Concept A Mars mission accommodation.

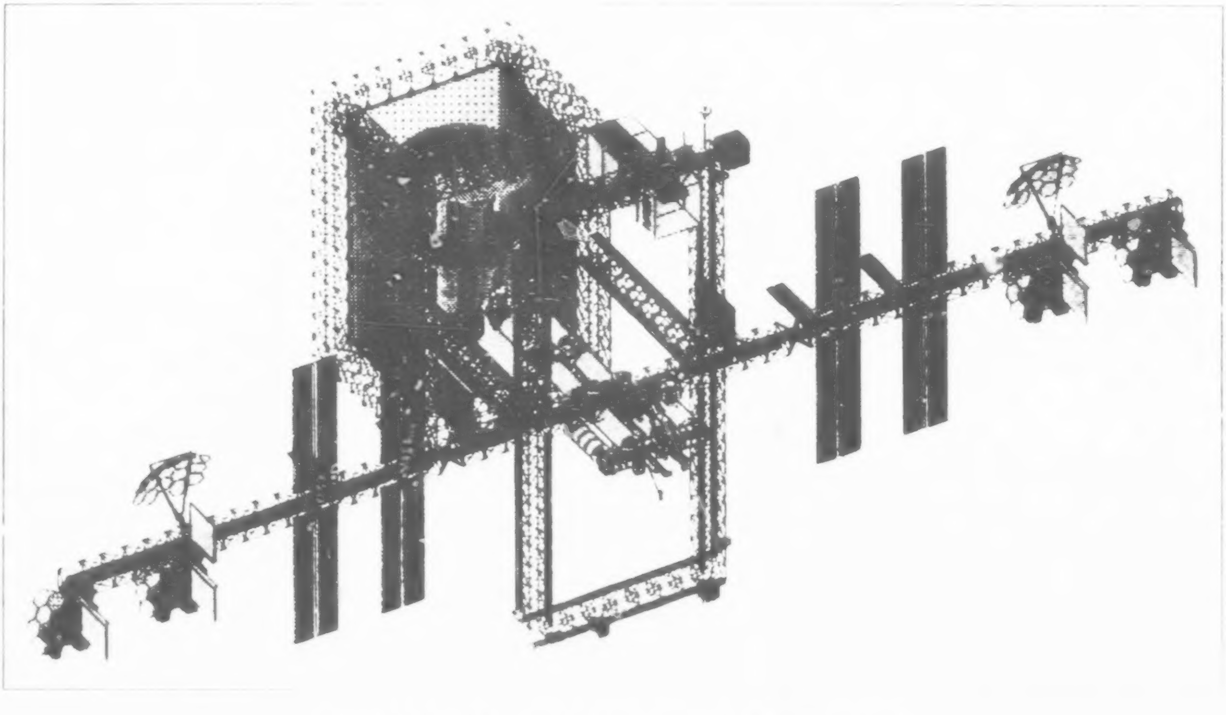

Figure 3.6.2-3.- Concept B Mars mission accommodation.

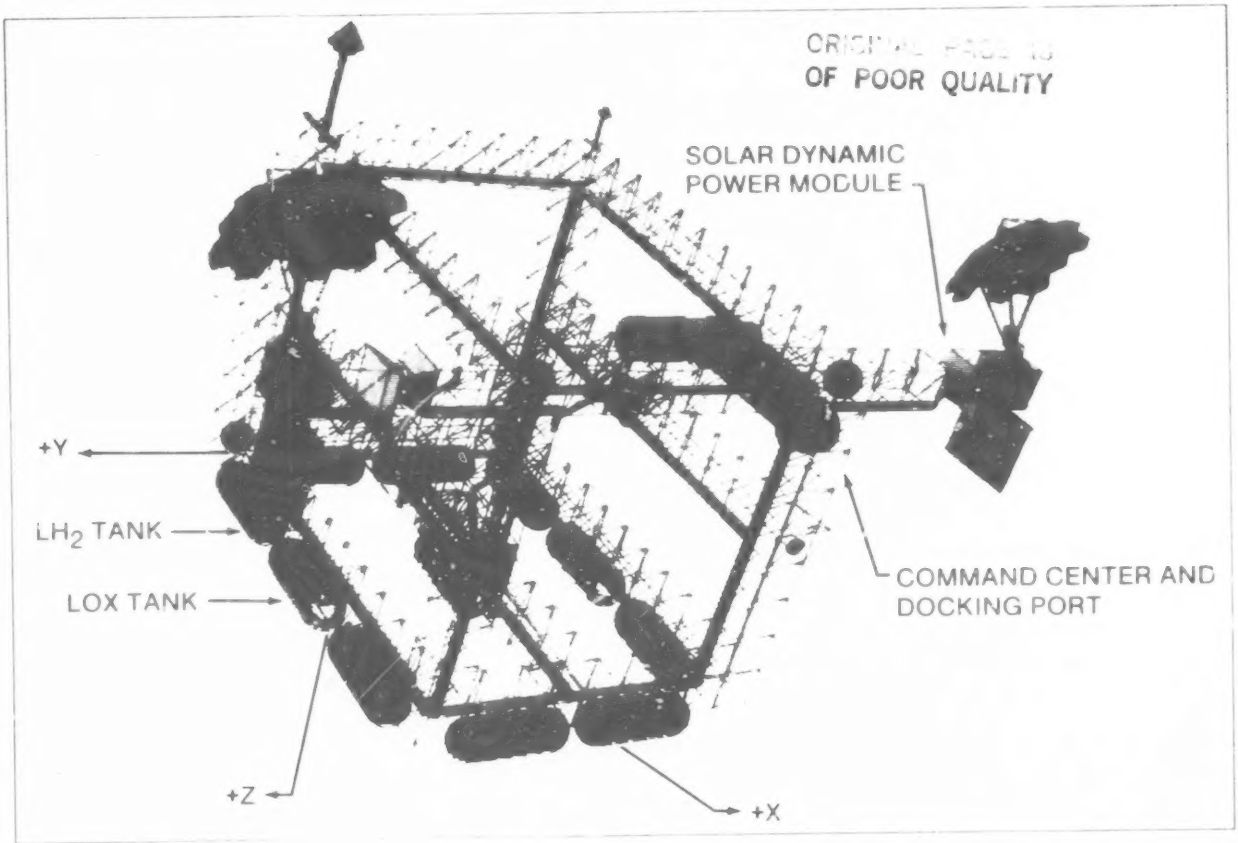

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Figure 3.6.2-4.- Man-tended transportation depot (open box concept).

certain amount of angular momentum is built up and countered with torques produced by the CMG's. When the momentum build up reaches a level near the CMG's torque-producing capacity, the station's RCS jets are fired in a way which releases the built-up angular momentum and allows the CMG's to return to a lower level of torque. This process is called desaturation of the CMG's.

The momentum buildup is divided into two distinct components, cyclic and secular. Cyclic momentum results from forces which grow and then dissipate through an orbit such that the net buildup is approximately zero. This type of momentum buildup is important because, even though the net value is negligible, the peak value is generally so large that it determines the size of momentum storage device needed. Secular momentum results from forces which vary in magnitude such that the net resulting momentum is nonzero. Values of secular momentum are generally lower than those for cyclic, but it is the secular component which must be dissipated periodically to avoid exceeding the capacity of the momentum storage system.

Along with the use of CMG's to store momentum, the station makes use of the fact that it is generally possible to maintain an attitude which minimizes the magnitude of the environmental forces. This minimum torque attitude is expressed as three ordered Euler angles, torque equilibrium angles (TEA's), which represent successive yaw, pitch, and roll rotations about the body axes.

Finally, since aerodynamic forces produce drag on the station in proportion to its total projected area, it is important to ensure that the addition of large hangars does not cause its orbit to decay too rapidly. Four parameters are useful for this purpose: the ballistic coefficient (M/CdA), the time it takes the station to decay from a 220-nmi orbit down to 150 nmi, the decay time from 150 nmi to reentry, and the amount of propellant required to perform a reboost back up to 220 nmi every 90 days.

Tables 3.6.2-VI through -VIII show the mass properties, TEA's and momentum values, and orbit decay parameters of each growth concept for Case Study 2.

The mass properties (table 3.6.2-VI) are shown for each concept, with and without vehicles, compared to the phase 1 station. Thus, total mass buildup can be seen, as well as the change in inertias and center of mass (CM) location. In particular, it is the CM location and the inertias which most affect the minimum torque attitude

TABLE 3.6.2-VI.- CASE STUDY 2 CONFIGURATION MASS PROPERTIES

Configuration	Mass (kg • 10³)	Center of mass (M)			Moment of inertia (kg • M²X 10⁹)			Product of inertia (kg • M²X 10⁶)		
		X	Y	Z	I_{xx}	I_{yy}	I_{zz}	I_{xy}	I_{xz}	I_{yz}
Phase 1 SS Freedom	219	-3.33	-0.02	3.69	1.93	2.31	3.24	1.12	-1.20	-0.55
Concept A	458	-3.61	-0.48	5.87	73.9	23.8	58.0	-0.86	-4.66	17.06
Concept A w/cargo veh.	689	-6.41	-0.29	20.26	106.26	57.47	60.68	-1.40	-67.63	19.75
Concept A w/pilot veh.	897	-7.74	-0.23	18.54	92.96	44.77	62.43	-1.72	-59.46	19.50
Concept B	470	-12.16	-0.99	1.01	65.38	31.98	74.36	-2.21	0.67	13.80
Concept B w/cargo veh.	701	-28.76	-0.66	0.67	67.02	72.85	115.0	-9.92	8.50	13.64
Concept B w//pilot veh.	909	-36.48	-0.50	-7.99	75.82	99.94	134.5	-13.83	213.3	95.17
Open box	118	-3.1	-19.5	-12.2	8.0	8.8	9.7	0.02	-2.9	-0.08
Open box w/pilot veh.	1,190	-22.7	-20.0	17.9	53.1	49.9	53.9	0.33	-10.9	-0.32

TABLE 3.6.2-VII.- CASE STUDY 2 CONFIGURATION CONTROLLABILITY ASSESSMENT

Configuration	Torque equilibrium attitude (deg)			Secular momentum	Peak cyclic momentum
	Yaw Ψ (Z)	Pitch Θ (Y)	Roll Φ (X)	N-M-S	N-M-SS
Phase 1 SS Freedom	1.0	-10.8	0.3	504	3,082 (Y)
Concept A	-0.5	-2.9	-2.9	5,875	10,034 (Y)
Concept A w/cargo veh.	-3.25	-7.64	-29.1	19,937	17,429 (Z)
Concept A w/pilot veh.	-0.5	10.82	-6.78	5,481	11,136 (Y)
Concept B	-0.75	-0.89	-1.86	2,861	14,310 (Y)
Concept B w/cargo veh.	-1.1	0.81	-1.94	5,002	24,410 (X)
Concept B w//pilot veh.	2.0	18.14	-1.95	9,935	19,018 (Y)
Open box	-2.6	36.6	1.3	467	5,800
Open box w/pilot veh.	-4.4	-44.4	0.0	2,335	3,270

and corresponding momentum buildup. In each station growth case, the CM location is measured with respect to the standard station axis system centered at the middle of the transverse boom, with X nominally along the velocity vector, Y extending out the starboard side, and Z along the nadir vector. For the open box transportation depot concept, the axis system is centered at the centroid of the starboard face, so its CM location appears somewhat different. All inertias are measured about the CM.

The controllability analysis (table 3.6.2-VII) show results for each concept, with and without vehicles, as compared to the phase 1 station. The concepts for Mars mission support undergo a significant change in both attitude and momentum buildup by the addition of hangars, vehicles, etc. The implication is simply that very different requirements are placed on the control and pointing systems, depending on the vehicle housed. This is due both to the large size of the Mars vehicle relative to that of the station, and the large projected areas which hangars and servicing facilities produce.

The orbit lifetime results (table 3.6.2-VIII) reveal that the combined addition of hangars and servicing bays (higher projected areas) to massive vehicles produces only moderate changes in the ballistic coefficient and orbit decay times. It should be noted, though, that projected area is a partial function of flight attitude, so that by changing attitude a somewhat different ballistic coefficient and decay time could be obtained. The most interesting result shown in this figure is the amount of propellant required to reboost the system back to 220 nmi every 90 days. This value ties together the amount of orbit decay experienced in 90 days (a function of ballistic coefficient) and the total amount of mass to which additional acceleration must be imparted. Mars concepts that include vehicles decay at approximately the same rate as the phase 1 station, but because their masses are so much greater, the net result is that they require more propellant. Note, finally, that the Mars concept A without the vehicles has such a low ballistic coefficient (twice the mass, but nearly three times the area) that it decays to the point of reentry before the 90-day reboost occurs.

In summary, the enhanced phase 1 station can accommodate the Mars mission defined in Case Study 2. An HLLV with a payload capability of at least 91 t to station orbit is required to support the mission. An OMV capable of handling the HLLV payload is required. Onorbit servicing, cryogenic propellant handling and storage, automation and telerobotics, and automated rendezvous and docking technology programs must be enhanced and accelerated.

From the standpoint of the assembly/checkout function, it is clear that a large unpressurized hangar is required for

TABLE 3.6.2-VIII.- CONFIGURATION ORBIT LIFETIME CHARACTERISTICS WITH A 2σ ATMOSPHERE

Configuration	Mass (kg • 10^3)	Area (M^2)	Ballistic Coeff (kg/M^2)	OrbitDecay time (days) 220-150 (nmi)	150-0 (nmi)	90 d orbit keep Propellant rqmt at ISP=320 s; h=220 nmi (kg)
Phase 1 SS Freedom June 2001	219	2,126	44.9	121	15	4,447
Concept A August 2001	458	5,913	33.7	71	11	—
Concept A w/cargo veh June 2003	689	6,654	45.0	123	16	13,349
Concept A w/pilot veh November 2004	897	6,692	58.3	265	25	6,366
Concept B August 2001	470	6,109	33.4	55	8	—
Concept B w/cargo veh June 2003	[illegible]	[illegible]	[illegible]	128	16	[illegible]
Concept B w/pilot veh November 2004	[illegible]	[illegible]	54.5	264	[illegible]	[illegible]

thermal/radiation/impact protection, debris containment, better robotic and EVA access, and better lighting and viewing. A command center is required and should be located near the middle of the Mars Vehicle (at buildup) for best viewing. Propellant tanks must be properly located on the station or they will greatly affect torque equilibrium angles, center-of-mass location, and demands on the control system.

3.6.2.3 Case Study 3

This case study, Lunar Observatory, is characterized by a single cargo mission and a single piloted mission in 2004 and 2005, followed by one piloted mission per year through 2014. Staytimes at the lunar surface are 14 days. The short trip times mean that a zero-g countermeasures program is not required for this case study.

Because of the much smaller dry masses associated with the lunar mission, the entire vehicle can be brought to orbit on a single HLLV launch, eliminating the need for onorbit assembly if expendable transit vehicles are used. Some degree of onorbit checkout will be required, however, since the vehicles must still be fueled onorbit. Thus the capability for propellant storage and handling at the LEO transportation node is required. If reusable transit vehicles are employed, full-up onorbit servicing is required. These transportation systems will return to the station with some residual propellants, creating the need for a wet/hazardous processing facility at the station.

One scenario for Space Station Freedom operations in support of this case study could be the following:

a. OMV retrieval of HLLV payload (dry vehicles)
b. Vehicle assembly (if required)
c. Checkout of all systems
d. OMV retrieval of HLLV payload (propellants)
e. Propellant transfer to lunar vehicle
f. Prestaging checkout
OMV moving of lunar vehicle to departure point
h. OMV retrieval of lunar vehicle element (if reusable)
i. Service/refurbishing of vehicle (if resuable)
jj. OMV staging of lunar crew return vehicle for disposal (if expendable)

With an HLLV launch frequency of six per year, the total processing, mission operations, and recovery can be accomplished in less than 90 days. Thus, even in the first 2 years of operations when there are two flights to the lunar surface, the station research and development (R&D) operations would be impacted for less than 180 days. It also appears that a three-person, single-shift operation is adequate for accommodation of Case Study 3.

Facilities required at the station are an unpressurized hangar sized to handle the lunar vehicles and OMV's and lunar and OMV propellant storage facilities. The hangar facility is sized at 30 m by 15 m by 15 m. The mass of the hangar and associated equipment if 15 t. The propellant storage facility (tanks) mass is 17 t.

Mass summaries of two concepts for station accommodation of the lunar mission are given in table 3.6.2-IX. The two configurations studies are illustrated in figures 3.6.2-5 and -6. Concept A is minimal additions to the phase 1 station. The hangar is located at the "scar" point for the servicing facility (a phase 2 station element). Total additional hardware is 175.6 t (most of this is propellant and tanks and the 2 OMV's). An additional hab module has been added to accommodate the assembly crew as well as transient crews on their way to the Moon. In concept C, we have used the dual keel configuration and located the hangar and propellant storage facilities to minimize movement of the station center of mass, thus minimizing the impact of lunar operations on the station microgravity research program (during quiescent periods). This requires a mass increase of about 20 t for truss bays and utility runs. With propellant and the lunar vehicles at the station, the total mass in orbit is about twice the mass of the phase 1 station. The choice of either of the two concepts of figures 3.6.2-5 and -6 as the preferred option

TABLE 3.6.2-IX.- CASE STUDY 3 SPACE STATION FREEDOM GROWTH HARDWARE TO ACCOMMODATE LUNAR OBSERVATORY

Component	Configuration			
	Concept A		Concept B	
	Num	Mass (kg)	Num	Mass (kg)
Truss bays	8	750	54	3,900
Utility bays	8	250	50	13,800
Solar dynamics	2	11,400	2	11,400
Habitat module	1	19,500	1	19,500
Node	2	8,700	2	8,700
propellant + tanks	6	113,000	6	113,000
Hangar + equipment	1	15,000	1	15,000
OVM (dry)	2	7,000	2	[illegible]
Total additional hardware		175,600		[illegible]
+ Phase I station 219,000 kg		394,600		[illegible]
+ Lunar cargo veh 34,700 kg OR		429,300		[illegible]
+ Lunar piloted vehicle 32,800 kg		427,400		[illegible]

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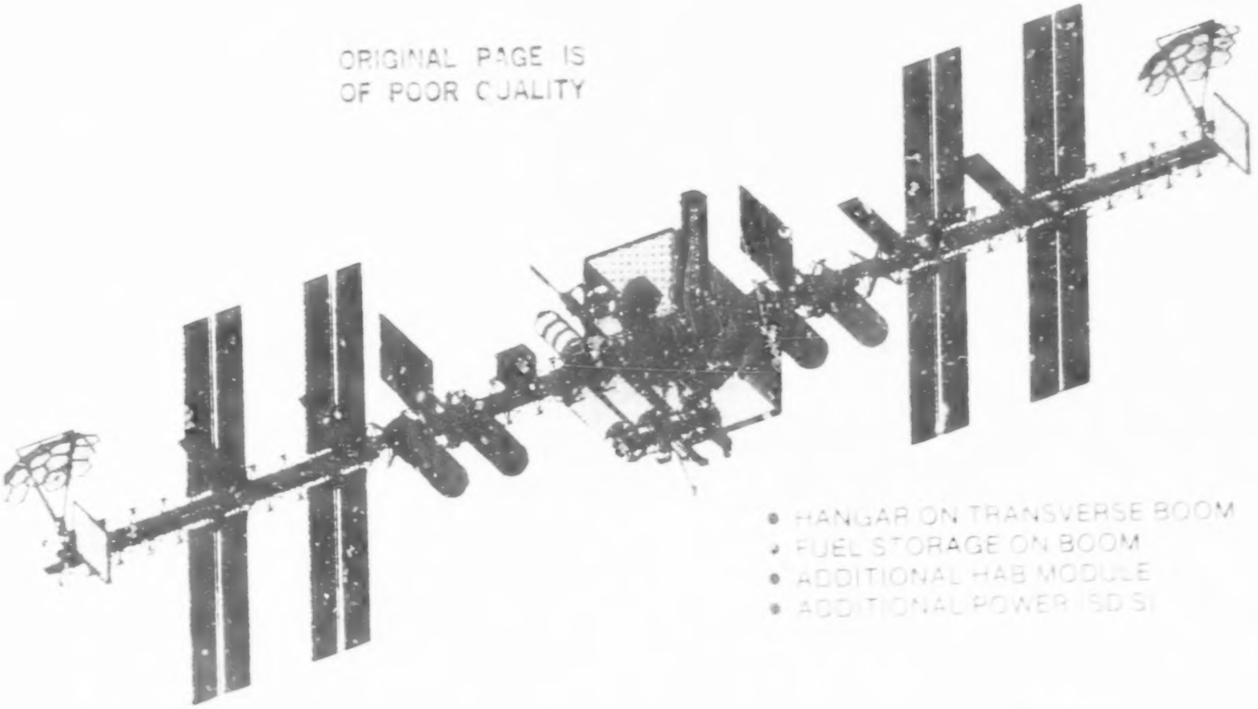
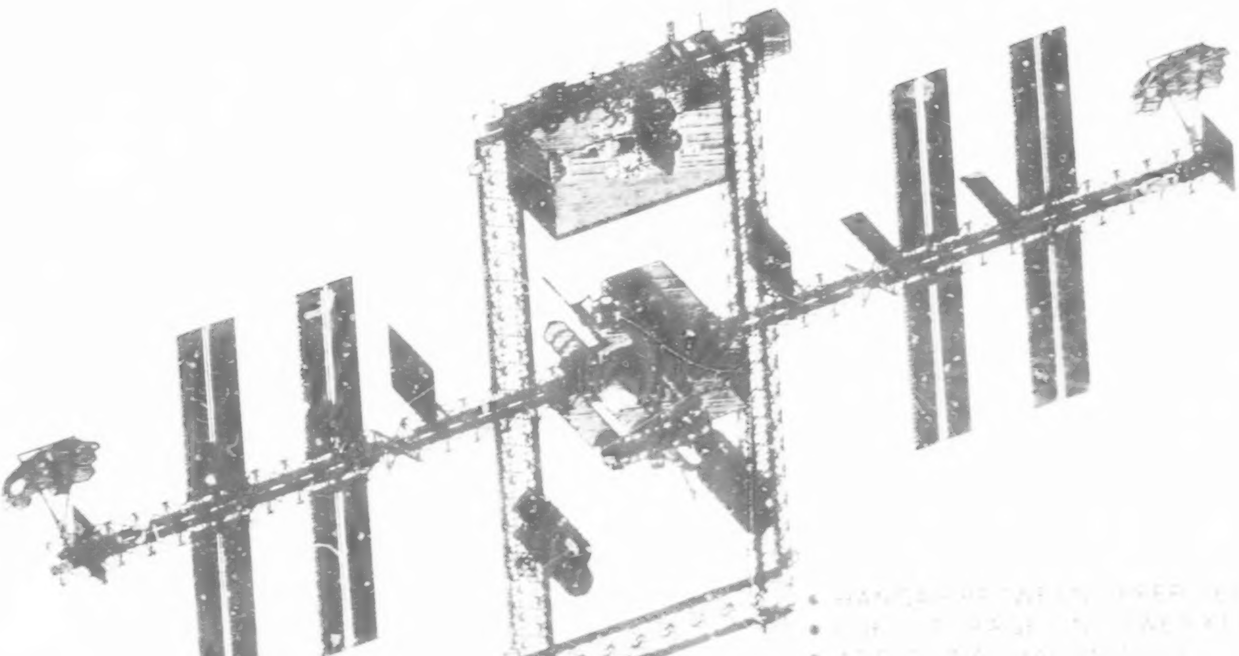


- HANGAR ON TRANSVERSE BOOM
- FUEL STORAGE ON BOOM
- ADDITIONAL HAB MODULE
- ADDITIONAL POWER (SDS)

Figure 3.6.2-5 - Concept A lunar observatory accommodation



- HANGAR BETWEEN UPPER KEELS
- FUEL STORAGE ON LOWER KEEL
- ADDITIONAL HAB MODULE
- ADDITIONAL POWER (SDS)

requires substantially more analysis, particularly in terms of the specific operations involved in preparation and checkout of the lunar vehicles.

The station growth concepts developed for Case Study 3 are similar to those developed for Case Study 2, although the vehicles and hangars are generally smaller and vehicle propellant is attached to the station. Thus, since they are subject to the same onorbit environmental disturbances, the analysis for the lunar mission support concepts consisted of the same set of parameters (TEA's, momentum buildup, orbital decay rates, etc.) outlined in Case Study 2.

Table 3.6.2-X shows the mass properties of each concept, with and without vehicles, compared to the phase 1 station. Clearly, the addition of vehicles, hangars, and propellant produces significant changes in the inertia characteristics, but because the propellant tanks were placed to balance the total system, the center of mass (CM) does not move as drastically as might be expected. As before, the coordinate system is centered at the middle bay of the transverse boom and inertias are measured about the CM.

The controllability characteristics of the lunar concepts (table 3.6.2-XI) show a somewhat smaller impact than the previous Mars concepts, due to the generally smaller vehicles. However, as witnessed by the large roll angles of concept C, even small vehicles cannot overcome big differences between the CM and the center of pressure (C P) produced by hangars, servicing bays, etc.

The orbital characteristics of the Case Study 3 concepts (table 3.6.2-XII) are much less dramatic. They have slightly higher ballistic coefficients, due mostly to the large mass and low area of propellant and tanks, and so have correspondingly longer decay times. Thus, although the lunar concepts have greater total mass than the phase 1 station, they require somewhat less propellant for reboost because they don't decay as far in 90 days.

In summary, the lunar observatory mission can be accommodated at Space Station Fredom with only modest additions to the station. The major infrastructure element which must be added is the space-based OMV capable of handling the HLLV payloads. If expendable lunar vehicles are used in conjunction with an HLLV, onorbit assembly may not be required, making the primary technology need cryogenic propellant storage and handling. With reusable vehicles, the mass to LEO can be reduced by 28 t the first year of operation, 36 t the second year, and 16 t in subsequent years. This savings is achieved at the cost of onorbit servicing of "wet" vehicles. Substantial study is required to establish the most efficient approach.

Finally, Case Study 3 has substantially less impact on the station base R&D activities. Thus there does not appear to be a need for a separate transportation node.

TABLE 3.6.2-X.- CASE STUDY 3 CONFIGURATION MASS PROPERTIES

Configuration	Mass (kg • 10^3)	Center of mass (M)			Moment of inertia (kg • $M^2 \times 10^7$)			Product of inertia (kg • $M^2 \times 10^6$)		
		X	Y	Z	I_{xx}	I_{yy}	I_{zz}	I_{xy}	I_{xz}	I_{yz}
Phase 1 SS Freedom	219	-3.33	-0.02	3.69	1.93	2.3	3.24	1.12	-1.20	-0.55
Concept A w/piloted veh.	427	-3.13	-0.56	4.77	32.8	4.4	35.1	11.7	-4.3	-6.6
Concept A w/cargo veh.	429	-3.29	-0.63	4.79	32.8	4.5	35.3	12.6	-4.7	-6.8
Concept C w/piloted veh. offset	446	-2.27	0.74	3.81	46.6	26.5	25.9	4.2	10.8	6.2
Concept C w/cargo veh. offset	448	-2.29	0.75	3.6	246.9	26.8	25.9	4.2	10.08	6.2
Concept C w/piloted veh. centered	446	-1.66	0.44	3.81	46.6	26.4	25.9	4.5	-1.2	12.2
Concept C w/cargo veh. centered	448	-1.65	0.44	3.62	46.9	26.8	25.9	4.5	-1.3	12.3

c - 3

TABLE 3.6.2-XI.- CASE STUDY 3 CONFIGURATION CONTROLLABILITY ASSESSMENT

Configuration	Torque equilibrium attitude (deg)			Secular momentum	Peak cyclic momentum
	Yaw Ψ(Z)	Pitch θ(Y)	Roll Φ(X)	N-M-S	N-M-S
Phase 1 space station	1.0	-10.8	0.3	504	3,082 (Y)
Concept A w/piloted veh.	2.8	-13.8	0.68	1,702	5,534 (Y)
Concept A w/cargo veh.	2.9	-13.8	0.68	1,702	5,534 (Y)
Concept C w/piloted veh.	-2.4	-1.5	31.0	8,877	8,233 (Z)
Concept C w/cargo veh.	-2.0	-1.5	26.6	7,656	6,895 (Z)
Concept C w/piloted veh.	-1.8	1.0	39.0	11,0621	0,637 (Z)
Concept C w/cargo veh. centered	-1.5	1.2	35.6	10,130	9,661 (Z)

TABLE 3.6.2-XII.- CASE STUDY 3 ORBITAL LIFETIME CHARACTERISTICS
WITH A 2σ ATMOSPHERE

Configuration	Mass (kg • 10^3)	Area (M^2)	Ballistic Coeff (kg/M^2)	Orbit Decay time (days)		90 d orbit keep Propellant rqmt at ISP=320 s; h=220 nmi (kg)
				220-150 (nmi)	150-0 (nmi)	
Phase 1 SS Freedom	219	2,126	44.9	121	15	4,447
Concept A w/piloted veh.	427	3,218	57.7	204	18	2,938
Concept A w/cargo veh.	429	3,230	57.7	205	20	3,061
Concept C w/pilot veh.	446	3,552	54.6	195	20	3,311
Concept C w/cargo veh.	448	3,572	54.5	195	20	3,319

3.6.2.4 Case Study 4

Case Study 4, Lunar Outpost to Early Mars Evolution, is still under development and is not addressed in this section.

3.6.2.5 Transportation Node Technology Requirements for all Case Studies

The technology drivers associated with the orbital node systems definition, and the attendant infrastructure requirements needed to support the "bold new initiatives" case studies, are discussed in the following paragraphs From an Earth orbital node viewpoint, it was found that the technology drivers identified in this study were relatively insensitive to the particular case study under analysis. The following discussion will apply to all of the case studies' technology drivers unless there was some issue unique to a particular case. Generally, need dates were the only major differences in the technology drivers for the four case studies, technology readiness requirements being keyed to the particular case study's program schedule.

For the LEO node systems definition, the required technologies, both enabling and enhancing, are those technologies concerned with providing the capability to assemble, process, and service the particular mission vehicle(s) in space. In this analysis, the node support requirements were divided into two categories, onorbit assembly and onorbit vehicle processing, so that onorbit resources could be quantified in terms of specific resources such as crew time, power, utilities, facilities, and supporting infrastructure. Depending upon the number and type of flights/reuse inherent in the case study mission design, this categorization allowed the in-space tasks to be classified further as either recurring or nonrecurring. This classification will be valuable in defining the LEO node and supporting systems requirements when additional in-depth studies are initiated. Cryogenic fluid management and autonomous rendezvous and docking are also discussed, but only from the standpoint of their effect on the onorbit operations.

A commitment to provide an extensive LEO node onorbit assembly and vehicle processing capability will require considerable future study effort. However, two important factors will undoubtedly influence the decision to provide this capability: the specific mission designs and the performance characteristics of Earth-to-orbit (ETO) launch systems.

With the exception of Case Study 1, Human Expedition to Phobos, all of the missions analyzed in the report included multiple missions with varying degrees of reusable mission elements. The ETO launch vehicles assumed in the studies were not capable of delivering fully assembled space vehicles to LEO, and the large aerocapture systems used extensively on the Mars and lunar transfer vehicles exceeded the payload volume envelopes. Therefore, the ability to process the mission elements required per flight and the need to service the resuable hardware drove the requirement for an onorbit processing capability. The launch vehicle payload mass and volume constraints drove the requirement for an onorbit assembly function at the node.

The major onorbit assembly and vehicle processing technology drivers/issues are shown in table 3.6.2-XIII. These issues will be addressed in terms of the two categories mentioned previously with comments on onorbit technology programs where appropriate.

TABLE 3.6.2-XIII.- ONORBIT ASSEMBLY AND VEHICLE PROCESSING TECHNOLOGY ISSUES

Mating/assembly - Large aeroshells - Spacecraft/propulsion stages - Telerobotic aids
Onorbit processing and handling of mission vehicles (including nuclear)
Space-based diagnostics/prognostics - Automated systems checkout - Automated systems health predictions/status
Cryogenic fluid management - Propellant transfer - Storage - Reliquefication
Onorbit processing of hazardous (wet) systems

To accommodate onorbit assembly at the LEO node, the capability to assemble, handle and mate/demate very large, very heavy, and complex space vehicles will be required. A high degree of confidence and reliability must be demonstrated and assembly operations must be conducted with minimum risks and minimum IVA/EVA crew involvement. For the planetary space vehicles (spacecraft, space propulsion systems) and any reusable elements/injection stages, the onorbit technology program must address handling, assembly and mating techniques using large capacity, highly articulated manipulators, and telerobotic/teleoperated aids. The success of providing the capabilities depends on major technological advances in the areas of automation and robotics, autonomous rendezvous and docking, and control of large structures.

In addition, the assembly operations associated with spacecraft aerobraking systems will require special handling and assembly techniques due to the close tolerance requirements and possible fragility inherent in the aeroshell thermal protection system (TPS). The onorbit technology program required to support the aerobraking systems must address the handling and assembly techniques associated with the joining of structural elements (welding, bonding, snap-connectors, etc.), the attachment/application of advanced TPS components/insulation and the removal and refurbishment of TPS materials and structural elements.

The onorbit vehicle processing function, while requiring many of the attributes needed by the orbital assembly function (i.e., handling, mating, manipulating large and massive mission elements in space) must also be capable of integrating, testing, and subsequent checkout of any and all elements of the space vehicle. To accomplish onorbit what has always been done using ground-based facilities will require a whole new set of operational philosophies, procedures, and support equipment — especially where manned systems are involved. Trade studies have been initiated to address these issues. However, from a technology standpoint, the orbital test programs for this function must focus on the development and implementation of advanced systems capable of performing automated checkout and systems status interrogations on each element as processed, and on the final flight configuration. In addition to the integration and checkout functions, the capability to service, maintain, and refurbish all reusable flight hardware elements (mission recurring operations) must be developed. Special consideration also must be given to the problems associated with processing hazardous (wet) systems that are used extensively on the reusable spacecraft.

In Case Study 4, the nuclear electric propulsion (NEP) cargo vehicle introduces some challenges to LEO node system definition that are more operational than technological. The projected orbital operations, which include the NEP vehicle assembly, processing, fueling, cargo loading, and periodic refurbishment/changeout of its thrusters, must be accomplished with minimum risks to the crew and the LEO node systems. Therefore, procedures and techniques must be developed that will ensure safe systems operations in and around the LEO node.

Without exception, the successful implementation of the case studies described in this report depends on efficiently managing cryogenic fluids in space. From the orbital node viewpoint, the capability to handle, transfer and manage large quantities of cryogenic propellants in space for long periods of time must be developed and demonstrated, onorbit, before these missions can seriously be considered. The facilities and techniques required to transfer the propellants from tank to tank and tank to vehicle, with minimum boiloff and contamination in and around the LEO node and mission vehicles, must be available early in the programs to be incorporated into the LEO node system definition and design.

Autonomous rendezvous and docking is another key technology driver in implementing the proposed case study missions. It is mandatory for Case Study 1, since this scenario assumes minimum or no supporting LEO node operations. Space-based systems must be developed that are capable of autonomous rendezvous and docking with very large, very heavy and passive vehicles such as expendable launch vehicles (ELV's), mission vehicles, and reusable transfer vehicles and injection stages. The system must be capable of stabilizing and maintaining control of these mission elements for subsequent handoff and transfer to the station, node, and/or coorbiting facilities with a high degree of accuracy. This capability must further be incorporated into an OMV-type system specifically tailored to handle large masses (<250 klbs) with adequate control authority to deliver and retrieve mission elements to and from staging orbits.

TABLE 3.6.2-XIV.- SUMMARY OF TECHNOLOGY FOR CASE STUDY TRANSPORTATION

- A LEO transportation node is required for Case Studies 2, 3, and 4.
- The phase 1 Space Station Freedom with currently planned scars can evolve to serve as the LEO transportation node.
- The amount of vehicle assembly/processing required versus ELV capability is highly scenario-dependent.
- Sizes and complexity of the mission vehicles dictate that modularity, high reliability, and telerobotic interfaces be incorporated into their designs.
- Space-storable cryogenics, autonomous rendezvous and docking, and automation and robotics technologies are enabling and must be accelerated.
- A space-based OMV-type system capable of handling and maneuvering masses in excess if 250,000 lbs is required.

Table 3.6.2-XIV summarizes the major findings of the Earth orbital systems definition analysis conducted on the FY 1988 exploration case studies. Further studies will be required, on a case-by-case basis, for all of the items listed in the figure before the orbital node system definition requirements can be completely developed. This is particularly true for Case Study 4, in which both lunar and Mars missions must be accommodated in addition to the assembly/servicing requirements imposed by a reusable NEP cargo vehicle.

3.6.3 Prerequisite Program Accommodation of Case Studies

The technical assessment of the case studies reveals a good degree of fit to the natural evolution of the Space Station Freedom program. The Transition Definition program has taken the next step to formulate preliminary plans to satisfy eventual accommodation of an exploration mission at the station. These plans address technical performance levels, schedule milestones, and budget requirements. As mentioned earlier, technical emphases will be on refinement of evolution concepts, definition and incorporation of baseline design accommodations, and development of technology readiness to enhance the baseline and enable evolution of the facility. The study plans in these areas are formulated to answer certain key questions fundamental to developing the station to serve as a transportation node.

3.6.3.1 Evolution Resource Requirements

The magnitude of effort needed to support exploration scenarios 2, 3, and 4 requires that the station increase the provision of its initial functions or capabilities. This expansion of the performance envelope includes such basic resources as crew, power, interior volume, and exterior workspace. The plan for accommodating the scenarios calls for using prerequisites provided by the OEXP to establish evolutionary station resource and facility requirements. These requirements were submitted as part of the baseline Preliminary Requirements Review (PRR) that was held in May and June of 1988. The broad envelope of requirements posed by exploration missions was shown in table 3.6.2-II. The ability to evolve to these levels of capabilities preserves the option to accommodate, at a top level of functionality, the three case studies employing a LEO node.

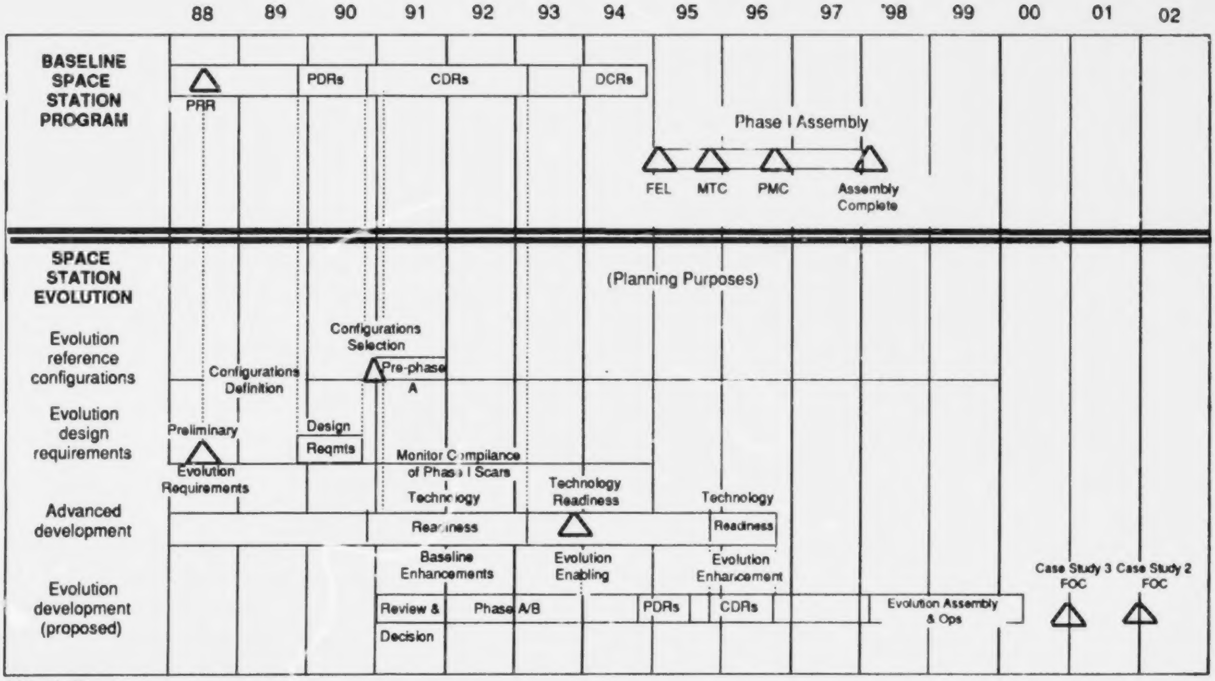
3.6.3.2 Baseline Design Accommodations

"Space Station Evolution - A Technical and Management Plan" embodies the policy for station evolution activities. As part of the planning function, the document discusses the self-imposed constraints on future infrastructure capability of not including provisions (hooks and scars) to enable evolution in the original design. Without derived engineering requirements on the systems and elements, the design will fail to provide the evolution performance levels. Thus, the baseline documentation does specify the transverse boom length and truss bay sizing necessary to add structure ensuring that the station can add the modules and growth facilities or hangars needed for vehicle processing and large structure assembly. The rotating alpha joints and radiator joints, located just inboard of the solar panels, are sized in documentation at levels exceeding the initial power and thermal requirements. The onorbit expense and operational complexity of changing out these units merit initial sizing to handle growth. Similarly, sizing scars are stated for the power and thermal distribution systems to preclude costly onorbit modifications. After further analysis, design requirements will be levied on the data management system so that future missions do not suffer from impaired command and control.

Schedule and Milestones

The milestone chart in figure 3.6.3-1 forms the basis of Transition Definition planning. The chart reflects the fact that the Transition Definition program is the bridge between ongoing NASA planning and technology development programs and the Space Station Freedom development program. All the evolution milestones for the next 4 years are tied to baseline program milestones.

The added functional capabilities required of a transportation node mean that additional baseline design provisions must be identified to enhance and enable evolution. In this instance, evolution from the baseline can occur onorbit as the addition of new hardware and/or the insertion of improved technology. The baseline Preliminary Design Review (PDR) is realistically the last opportunity to impact the design; therefore, system impacts must be well understood by the time it occurs. The PDR is distributed over nearly a year's time, beginning in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BASELINE SPACE STATION PROGRAM
PRR
PDRs
CDRs
DCRs
Phase I Assembly
FEL
MTC
PMC
Assembly Complete
SPACE STATION EVOLUTION
(Planning Purposes)
Evolution reference configurations
Configurations Definition
Configurations Selection
Pre-phase A
Evolution design requirements
Preliminary
Evolution Requirements
Design Reqmts
Monitor Compilance of Phase I Scars
Advanced development
Technology Readiness
Technology Readiness
Technology Readiness
Evolution development (proposed)
Baseline Enhancements
Evolution Enabling
Evolution Enhancement
Review & Decision
Phase A/B
PDRs
CDRs
Evolution Assembly & Ops
Case Study 3 FOC
Case Study 2 FOC

FOC = FullOperations Capability

Figure 3.6.3-1.- Space station evolution milestones.

November of 1989. The following are key areas of effort to identify evolution performance levels and design accommodation requirements:

a. Logistics systems evolution
b. Space transfer vehicle accommodation
c. Onorbit assembly/servicing
d. Advanced robotics for in-space vehicle processing
e. Advanced automation for in-space vehicle processing
f. Onorbit fluid management
g. Separate tasks to look at the impact on each distributed system of accommodating exploration initiatives

The integrated results from these analyses will permit assessment of the unique scenario implications posed by case studies then under review.

The Transition Definition program will continue to refine reference evolution concepts in anticipation of a review and decision on an exploration mission in the 1991 timeframe. Pre-phase A work will be performed in several concepts until 1991, at which time phase A studies could begin to determine concept feasibility and rough cost estimates on one or two candidate configurations. As part of the pre-phase A's, OSS and OSSA will jointly study concepts for a man-rated artificial gravity facility.

Initial concepts represent time-phased sets of functional capabilities, system performance, and resources that are viable means of case study accommodation. The concepts are derived by performing systems and operations analyses on the input requirements provided by OEXP. Results of these analyses will provide the following data: physical characteristics by accommodation option; operational characteristics, including assembly/checkout resource timelines, servicing facility timelines, and precursor research accommodation; configurations subjected to control and structural dynamics analyses; logistics and cargo-carrying requirements; hooks and scars on the baseline facility; and technology needs.

3.6.4 Support Required from Other NASA Organizations

3.6.4.1 Technology Requirements

The actual requirements for technology will be refined as the definition and preliminary design of the evolution configuration(s) emerges. The evolution advanced development program will be strongly coupled with the evolution system studies and analysis activities. A set of advanced technology needs will be identified based on the evolution mission and user requirements. The technology needs will be the driver for initial development activities and for a full-scale advanced program that will ensure the readiness of technologies that enable evolution.

The OSS advanced development program will thus be tied to OAST technology activities. The initial development of technologies based on station technology needs will be conducted by OSS and/or OAST, depending in part on the uniqueness of the technology to the station.

The near-term emphasis on automation and robotics technologies with high potential for near-term application is predicated on readiness before or during the Critical Design Review (CDR). Some of the disciplines requiring long-term development are identified in the section on transportation node technology requirements. These enabling technologies must be matured by the end of the evolution phase B studies. In addition, the necessary technologies will be developed to enhance the productivity of the station next century. The enhancing activities must be completed before or during the evolution CDR timeframe (approximately 1996).

3.6.4.2 ETO Support Requirements

The fundamental ETO support required by Space Station Freedom to serve as a transportation node can be divided into three groups:

a. Transportation of additional hardware elements (hangars, habs, etc.)
b. Crew rotation
c. Logistics support

In group a, the requirement is approximately 250 t for Case Study 2 and 74 t for Case Study 3. In group b, the requirement is to accommodate a total crew of 18 to 24 for Case Study 2 and 10 for Case Study 3. In both cases crew rotation cycle is every 180 days. Group c, logistics support, is not sufficiently well defined at this time to provide specific requirements. This group includes the vehicles and propellants required for the missions of the case studies and also includes the normal logistics support for the station.

3.6.5 Summary

Using exploration preliminary requirements, the Space Station Freedom program has assessed the feasibility of supporting the FY 1988 case studies. Preliminary analysis shows that there is a good degree of fit between Case Studies 2 and 3 and the natural evolution of the station. A technical assessment of the evolutionary strategy, Case Study 4, will be performed when that scenario is sufficiently defined.

The phase 1 station, with currently planned hooks, scars, and modular configuration, can evolve to serve as the LEO transportation node. The station program schedule is consistent with LEO node milestone requirements of exploration missions. Preliminary estimates have been provided for near-term cost impacts associated with supporting the exploration missions.

Station evolution planning mechanisms have been more thoroughly integrated with Agency exploration planning this year and can now be turned to refining the ability of the program to accommodate an exploration mission. With the provision of more detailed performance requirements by case study, it will be possible to impact station systems more precisely. This in turn allows iteration on design requirements and interfaces with other infrastructure providers. Additional technologies requiring accelerated development will be identified.

The Space Station Freedom program will be ready to support the review and decision on extending human presence into the solar system.

3.7 CASE STUDY PROGRAM INTEGRATION SCHEDULES

To meet its objective of providing a focused program of human exploration of the solar system, the Office of Exploration (OEXP) must coordinate the implementation of mission and system requirements within the appropriate NASA program offices. In particular, OEXP must ensure that certain program-level requirements are met, including

a. Technology needs, so that development on new space systems and mission capabilities may proceed in an orderly manner

b. Precursor science missions, to provide an accurate characterization of extraterrestrial bodies for system design purposes

c. Life science programs, to protect the health, safety, and well-being of humans in space

d. Earth-to-orbit (ETO) transportation and Space Station Freedom capabilities, to enable delivery and in-space assembly of the massive space systems required for these missions

e. Communications and tracking systems, to maintain continuous contact with the manned spacecraft

To assist in this process, program integration schedules were developed to provide a framework for integrating case study requirements with current planning for future technology, for space and life science, and for ETO and LEO systems development. The intent of the schedules is to identify the implications and consequences of each case study with respect to current and near-term programs, so that the requirements can be accommodated in these and future programs. In some cases during the iterative synthesis process, it was found that technologies or systems required to support the case studies were not available within the proposed timeframe. Typically, the problem is either budgetary (i.e., precursor science missions or technology advances are delayed because of low levels of funding) or technical (i.e., identified prerequisite requires a technological breakthrough). Therefore, the technology development programs were expanded or accelerated, or the mission timelines were slipped to a time scale consistent with the respective development capabilities.

These schedules represent a preliminary attempt to develop an overall case study program plan. Although iterative efforts were made to merge the case study needs with the NASA program offices' plans, a few remaining issues that could not be resolved easily will be addressed in next year's activities.

The program integration schedule for each case study is in two parts. The first part of the schedule for each case study describes mission milestones for timelines and development schedules for space transfer vehicles, surface systems, and nodes. Prerequisite requirements for the case study are given in the second part of the schedule. For technology needs, the triangle indicates the date the technology is required and the number inside the triangle indicates the required technology readiness level.

In the mission milestones part of the schedule, the development schedules are presented at the top level. Shown are dates for definition and conceptual studies (phase A/B), design and development (phase C/D), delivery of the first production unit, flight test program, assembly and checkout, and initial operational capability (IOC). These milestones are shown as inverted (point down) triangles with the number inside indicating the month of the event.

The length of the development phase was defined in the Exploration Requirements Document (ERD) for each case study. The end of phase C/D was defined here to coincide with the delivery of the first production unit to NASA. The schedules of all newly developed flight vehicles included a 1- to 2-year flight test program assumed to consist of launch to low-Earth orbit (LEO), assembly, and flight test. It was further assumed that the assembly and checkout activities would be performed in conjunction with the flight test program. Surface system IOC dates were defined to occur 1 year before launch to permit prior integration of the system into the vehicle.

Program schedules and major prerequisite need dates are depicted in the Prerequisite Requirements portion of each schedule. To focus the effort according to the NASA organization structure, prerequisite needs are classed as launch vehicles, science precursor missions, life science needs, and Space Station Freedom usage. These needs are presented in terms of design requirements, methodologies, technology status/demonstrations, and systems. Detailed program plans to support the defined need dates will be provided by codes M, E, R, T, and S.

Schedules for each of the four case studies are provided in figures 3.7-1 through 3.7-4. Each of the schedules is described below in terms of its mission and prerequisite requirements.

3.7.1 Case Study 1

The program integration schedule for Case Study 1 is shown in figure 3.7-1 and described in detail in section 2.1.

The ERD specifies that the Phobos mission is to have an accelerated development cycle in accord with its objective to send the first humans to the martian system. Therefore, the schedules for Case Study 1 assume a 5-year phase C/D period, typical of the Apollo program, for all vehicle and surface systems except the piloted vehicle, which assumes a 6-year period. Phase A/B, for this and most other case studies, was assumed to require a 3-year period.

Principal prerequisite requirements for Case Study 1 include

a. Zero-gravity physiological research to verify human health and performance
b. A Mars orbiter mission to provide mapping and site selection data for the teleoperated science rovers
c. Knowledge of Phobos surface properties for extravehicular activity and vehicle systems and operations

3.7.2 Case Study 2

The program integration schedule for Case Study 2 is shown in figure 3.7-2 and detailed in section 2.2.

Case Study 2 assumes aggressive 5-year development programs for the vehicles and a longer period for the exploration systems. This is consistent with its objective to establish the first human presence on another planet at an early date.

Principal prerequisite requirements for Case Study 2 include

a. Zero-gravity physiological research and countermeasures development in support of long-duration space flight
b. Mars orbiter as well as teleoperated rover and sample return missions to validate surface/environmental characteristics in support of human safety; EVA and vehicle development; and site selection for science activities
c. Development of high-Isp engines
d. Demonstration of aerocapture at Mars and Earth orbits

3.7.3 Case Study 3

The program integrated schedule for Case Study 3 is given in figure 3.7-3 and described in detail in section 2.3.

This case study was assumed to have a 6-year development schedule, which corresponds to its objective of establishing a long-term science outpost on the Moon at a moderate pace.

Principal prerequisite requirements for Case Study 3 include

a. Global lunar mapping and local site surveys to provide detailed information for site selection
b. Demonstration of aerobraking at Earth orbit

3.7.4 Case Study 4

The program integration schedule for Case Study 4 is shown in figure 3.7-4 and described in section 2.4.

Cases Study 4 uses nominal 5- to 6-year development schedules and phase A/B was specified as a 3-year period. However, the duration of phase A/B for the electric cargo vehicle was estimated at 4 years to provide additional time for feasibility analysis and power source definition, since high-power electric propulsion systems were not considered as advanced as comparable chemical propulsion systems.

Because of the complexity of this case study, there are a large number of precursor requirements, including

a. Global resource mapping of the Moon and local site surveys to select and verify a site for resource processing
b. Global Mars mapping and sample return missions to provide data on the martian surface and environment to support site selection
c. Sample return from Phobos to determine the surface properties and composition in support of proposed propellant production activities
d. Megawatt power capability in support of the electric cargo vehicle
e. Long-duration life support for extended human missions
f. Aerocapture demonstrations for Mars and Earth orbit application
g. Lunar and Phobos propellant-processing feasibility demonstrations

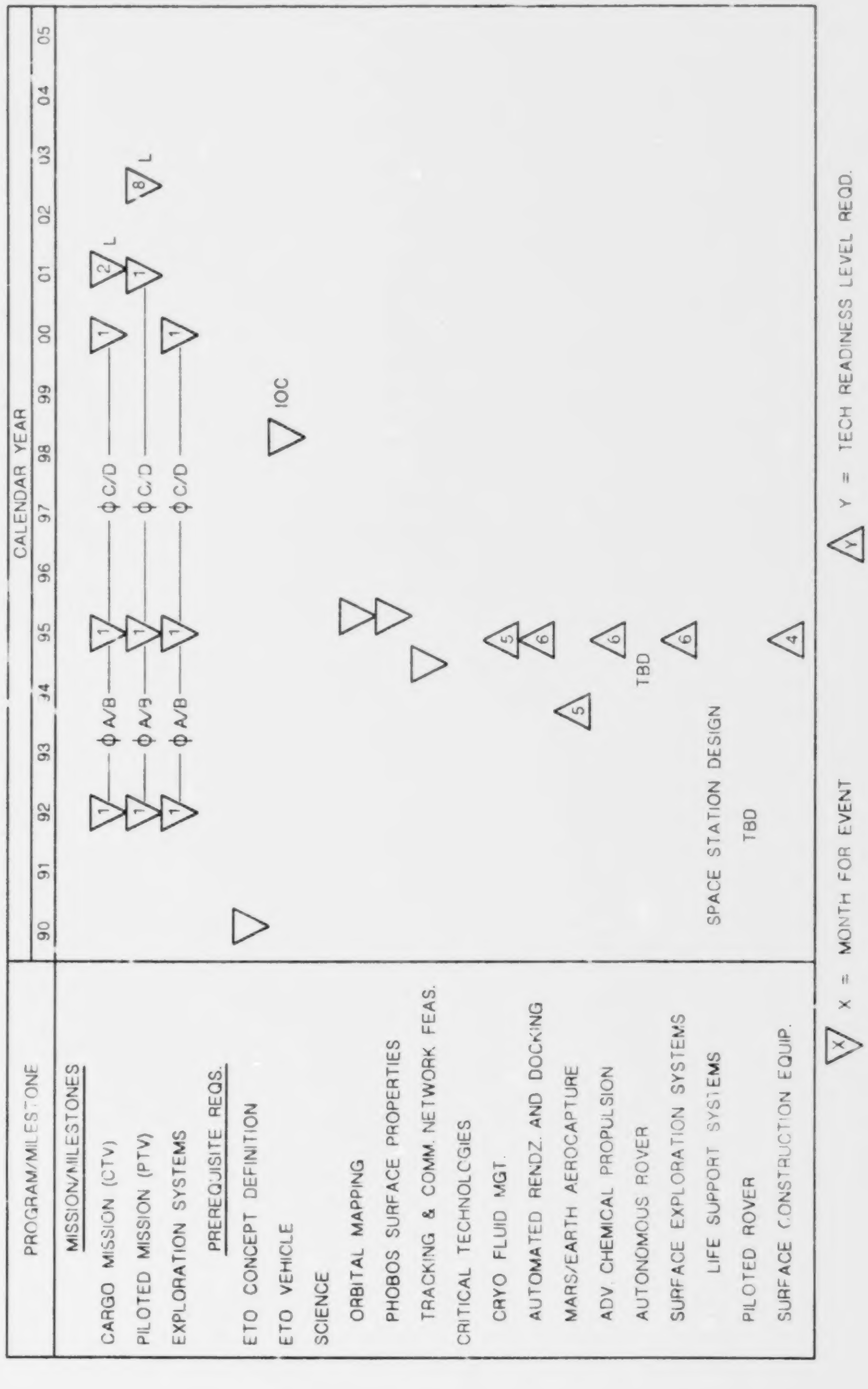

Figure 3.7-1.- Program integration schedule — Human Expedition to Phobos.

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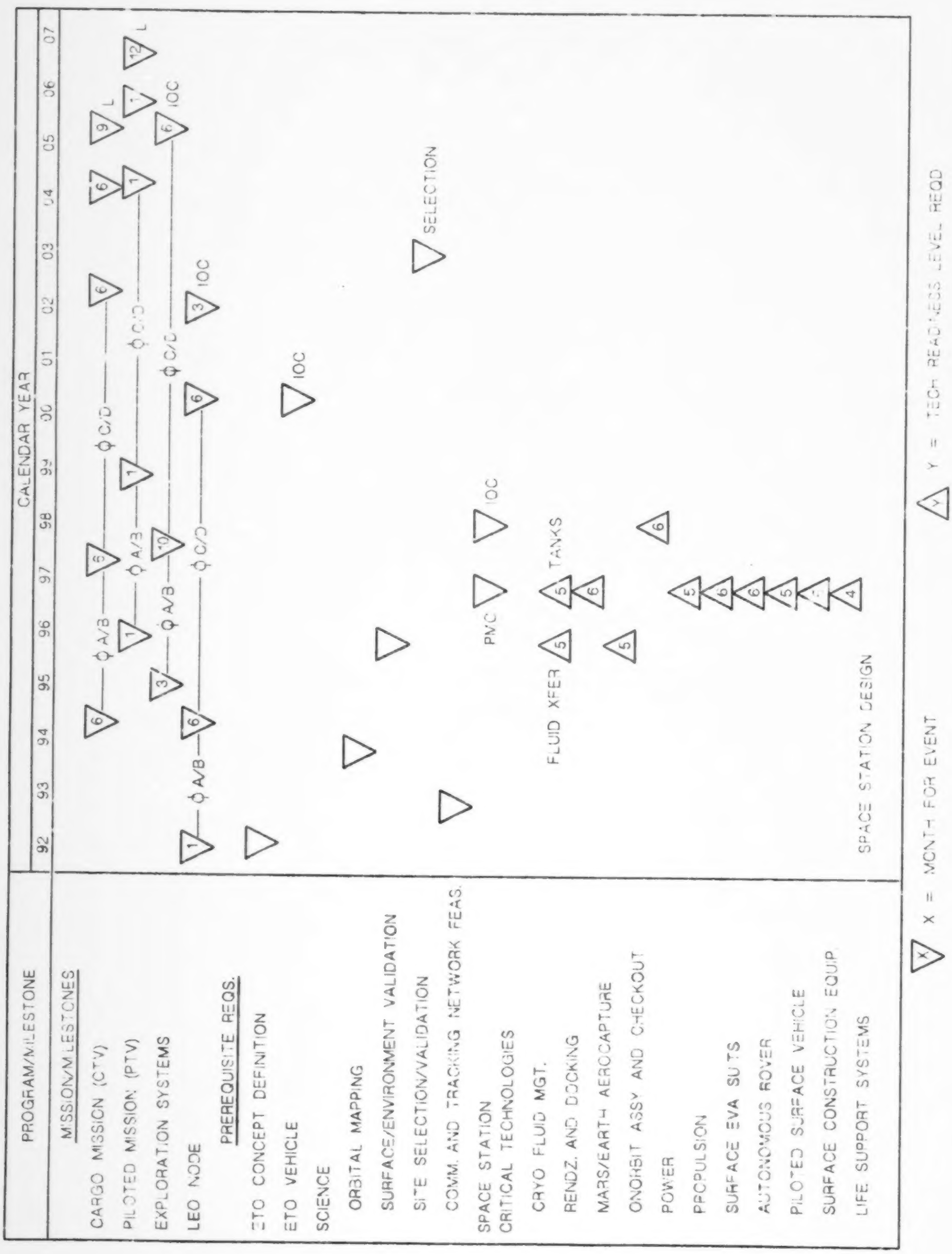

Figure 3.7-2.- Program integration schedule — Human Expeditions to Mars.

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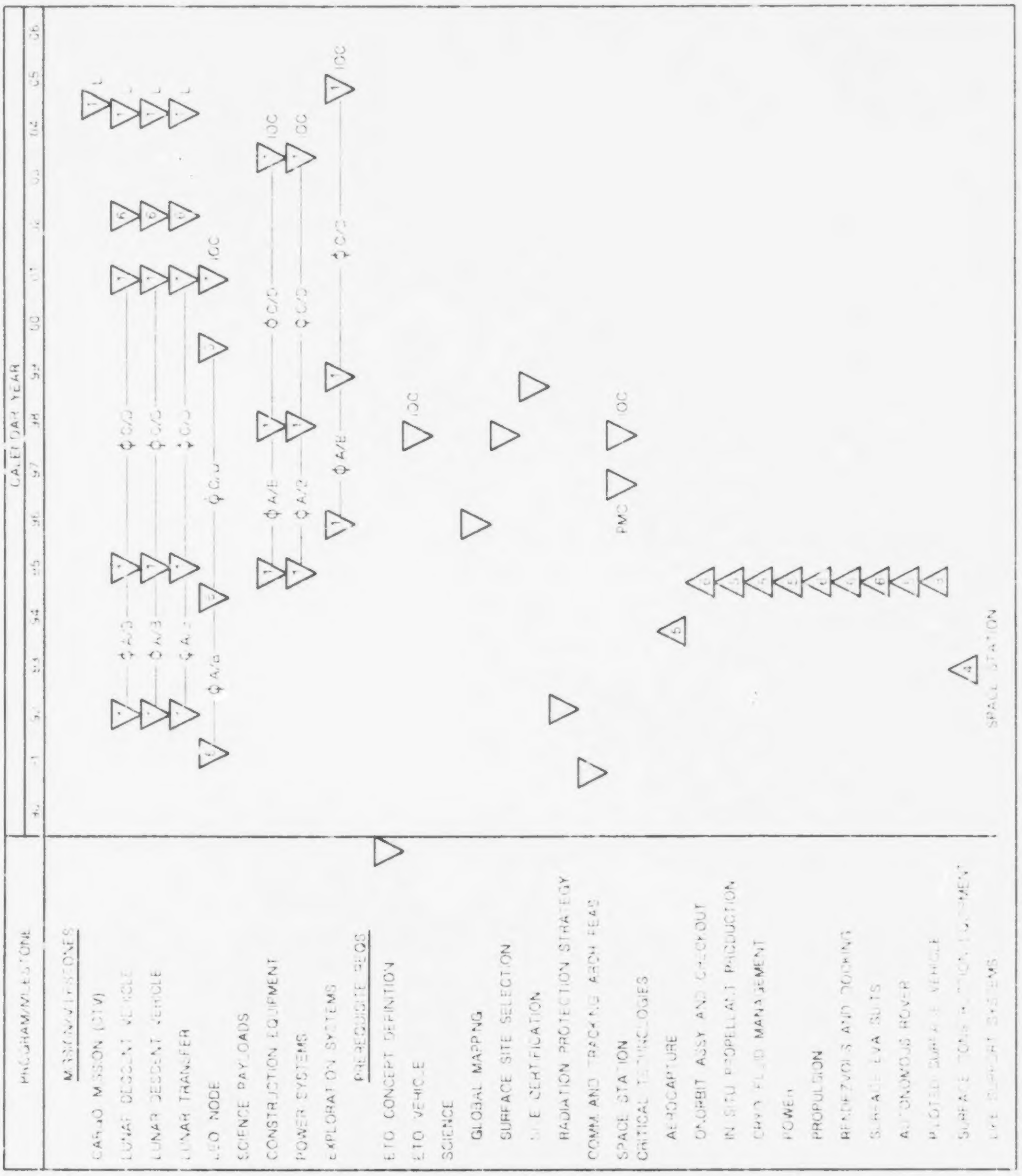

Figure 3.7-3.- Program integration schedule — Lunar Observatory.

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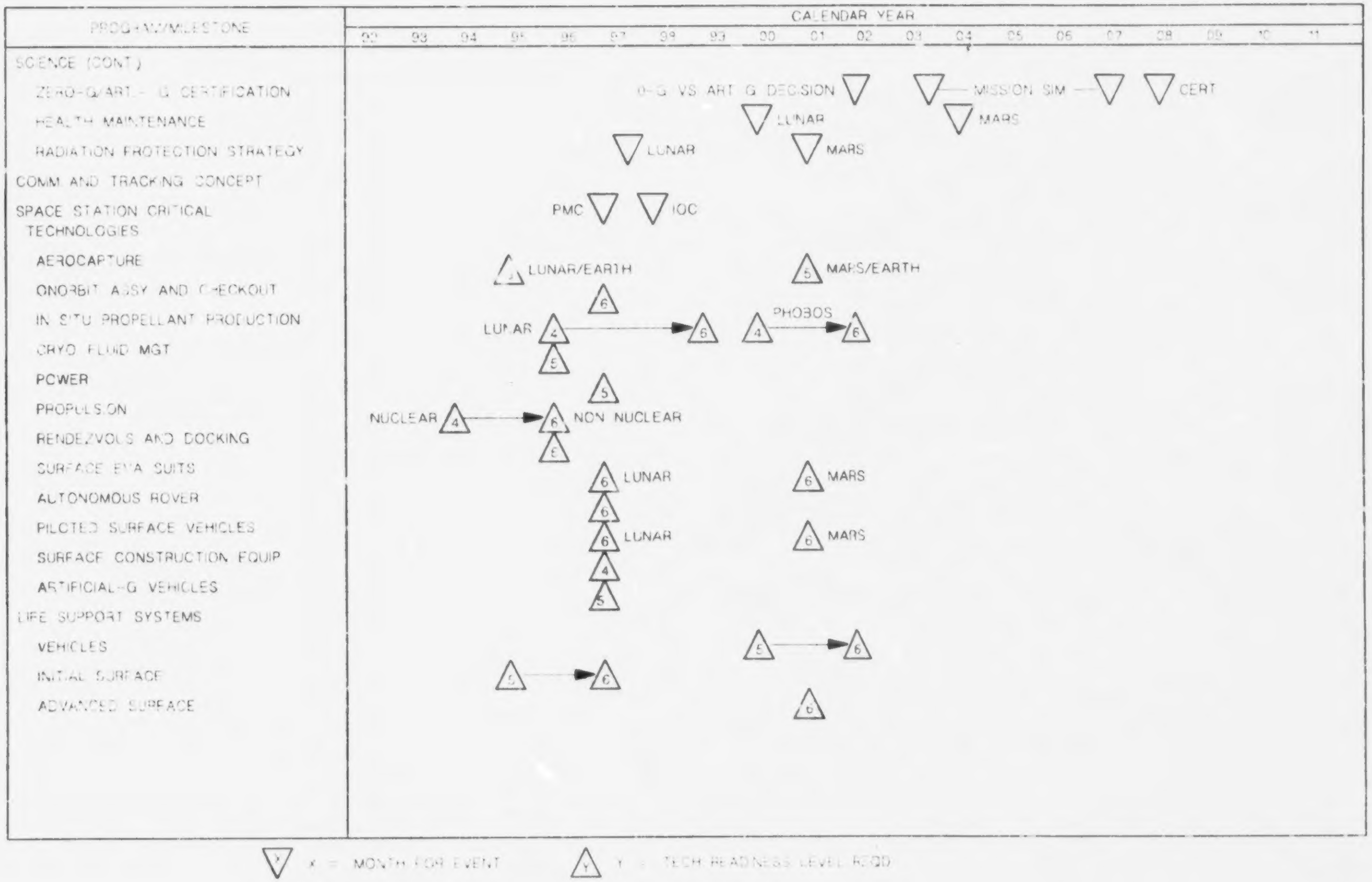

Figure 3.7-4.- Program integration schedule —
Lunar Outpost to Early Mars Evolution.

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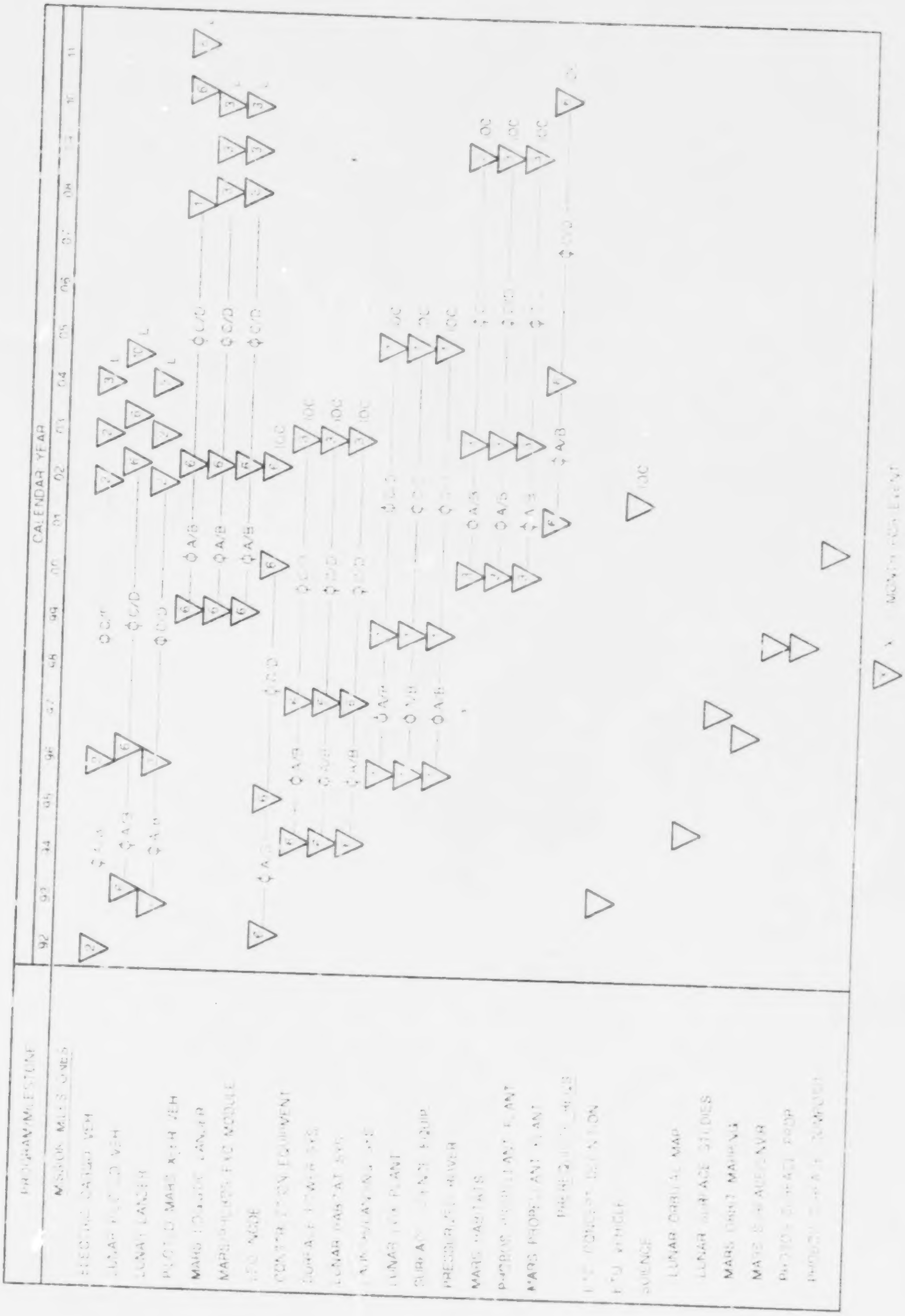


Figure 3.7-4.- Concluded.

SECTION 4

Special Reports and Studies

In addition to the systems definition studies performed by the Integration Agents (IA's) in direct response to the specific case study requirements, the Office of Exploration (OEXP) activities also include studies and product development that warrant special focus and emphasis. These studies and activities fall within three identifiable categories: broad mission and system architectural trades, system and technology oversight and definition support, and programmatic special reports.

a. Broad trade studies (sections 4.1-4.4) are studies that affect more than one case study, are case-study independent, or involve multiple integration areas. The criteria for inclusion in the FY 1988 study cycle were that the results either be essential to further mature the case studies during FY 1989 or enable a technical "assault" upon constraints to the case study FY 1988 implementation.

b. System and technology oversight and definition support is provided by technologists in selected areas through the conduct of discipline technology workshops and through tasks in direct support of the IA's (sections 4.5-4.8 and 4.11-4.14).

c. Programmatic special reports (sections 4.9, 4.10, and 4.15) generally relate to specific topics that are established and produced at the direction of OEXP.

The sections cited above provide results and highlights of tasks that have been initiated, are continuing, or have been completed during FY 1988. Table 4-I summarizes the major results of each study.

TABLE 4-I.- SPECIAL TRADE STUDIES AND REPORTS SUMMARY

Study/report	Affected Case Study	Objective	Results
4.1 Earth-Moon Node Location	3, 4	Case Study 3 Define preferred location (LEO or LLO) for the Case Study 3 transportation node.	Case Study 3 (1) A LEO node is highly desirable if not mandatory. (2) An LLO node is not justified at this time.
		Case Study 4 Determine the most desirable location in Earth-Moon space for supporting the transportation needs of the Mars portion of Case Study 4.	Case Study 4 (1) For steady-state operations, locations away from LEO are preferable if LLOX is to be used. (2) Conclusions cannot be drawn at this time as considerations other than mass to LEO are influential.
4.2 Extraterrestrial Propellant Leveraging	4	Determine the impact and potential benefit of augmenting mission objectives with extraterrestrial (lunar, Phobos/Deimos, or Mars) resources.	(1) Preliminary assessment is that Phobos/Deimos propellant for homeward leg offers more leverage for chemical propulsion system than LLOX for outbound leg to Mars.

TABLE 4-1.- Continued

Study/report	Affected Case Study	Objective	Results
4.2 (cont.)			(2) Phobos/Deimos propellant may be beneficially exported to LEO while export of LLOX to LEO via chemical propulsion systems may not be beneficial.
			(3) Buildup and payback time will be important constituents of the benefit.
4.3 Feasibility of Automating LLOX Production	3, 4	(1) Prove the possibility for an automatic LLOX production site.	(1) Required material handling capacity of the mining and transport equipment is dependent upon the particular O_2 extraction process; automation and robotics is only one of several trade-off considerations.
		(2) Evaluate the feasibility of automating the specific processes being assumed.	(2) Except for specific adaptation to lunar conditions, the required automation and robotic technology is the kind currently used on Earth. Mining and transport functions with minimum human attendance should be possible.
4.4 Low-Earth-Orbit Assembly Strategy	1, 2, 3, 4	(1) Identify studies and interactions needed to develop and ensure end-to-end assembly capability as required for each case study.	(1) The major unresolved issue is how much emphasis to place on each element of the infrastructure; e.g., ETO, space transfer vehicles, transportation node, and operations.
		(2) Evaluate the assembly of Case Study 1 (Phobos) vehicles.	(2) ETO capability of between 230-250 t should be considered.
		(3) Determine spacecraft assembly, fueling, and verification onorbit tasks and establish the method (EVA, automation, etc.) for performing these tasks.	(3) There is concern whether the Phobos vehicles as "designed" can be mated without an onorbit node.

TABLE 4-1.- Continued

Study/report	Affected Case Study	Objective	Results
4.4 (cont)		(4) Assess the operational methods and techniques for LEO assembly.	(4) Space transfer vehicles should be designed with assembly as a design criteria. (5) Current vehicle processing flows as performed on the ground are not viable for onorbit and significant changes will be required, including designing the proper features into the space vehicles, using AI and robotics as much as possible, and performing functions prelaunch as much as possible.
4.5 Power and Propulsion Parameters for Nuclear Electric Vehicles	4	Provide a common and consistent set of projected nuclear propulsion vehicle performance parameters for use in all relevant case studies.	A data base of thruster performance and reactor specific mass has been established and has the concurrence of OAST
4.6 Teleoperated Rovers in Support of Human Planetary Exploration	1, 2	Examine feasibility and utility of controlling Mars rovers from martian orbit.	Case Study 1 (1) Available crew hours at Mars are insufficient to take advantage of crew presence for teleoperation if rover teleoperation window is constrained to the period when crew is in Mars orbit and is not extended to doing advanced operations from Earth. (2) Planning for rover teleoperation rather than for a level of rover autonomy appears to require further depth of assessment. Case Study 2 (1) Rover devices are expected to add to the operational safety and effectiveness of crews on a planetary surface as well as serving as scientific tools.

TABLE 4-1.- Continued

Study/report	Affected Case Study	Objective	Results
4.7 Lunar Observatory Staytime Extension Study	3	Determine mass of advanced solar-based power system to enable a crew staytime extension into the night during construction phase of Case Study 3.	(1) A regenerative fuel cell capable of supplying day/night power for multiple lunar nights is lunar estimated to have a 7-8 t mass. This capability could support any future requirement to reduce the number of construction flights as well as potentially augmenting the operational-phase power system.
4.8 Phobos Exploration Assessment	1, 2, 4	Develop an understanding of the problem of remaining on the surface of Phobos under its milli-g environment, and flying over the surface as a mobility option for translating from point to point on Phobos.	(1) The potential for leaving the surface for extended periods (10-30 min) due to inadvertant reaction forces is high, seemingly indicating a need for a method to anchor to the work site. (2) Flying over Phobos gains the benefit of ease of traverse at the expense of systems and consumables required for controlled flight.
4.9 Space Exploration Cost Understanding	1, 2, 3, 4	Conduct an indepth assessment of the costing methodology for the purpose of updating the assumptions and art of costing major initiative concepts whose projected implementation is to occur far into the future.	NASA culture has changed in the last 2 decades; however, additional changes should be made. Recommended changes are: (1) Acquisition cost realism along with unit production cost as a significant design requirement (2) Planned product improvements, and maximum use of proven components and subsystems (3) Presence of a continuous alternative (4) Short and stable schedules for development and production

TABLE 4-1.- Continued

Study/report	Affected Case Study	Objective	Results
4.9 (cont.)			(5) Experienced, small staffs with clear command channels and limited reporting (6) Effective communication with users for cost/performance trade- offs (7) Early development-phase funding for production and support considerations (3) Use of mass production techniques as much as possible (9) Technology advancement at reasonable rates (10) Minimum functional complexity of individual hardware elements (11) Commonality among hardware elements (12) Hardware designed with substantial performance margins
4.10 Science Opportunities in Human Exploration Initiatives	1, 2, 3, 4	Provide an initial assessment of the science opportunities in human expedition missions and begin laying the groundwork for developing and integrating science requirements into the case studies.	The major accomplishments have been associated with the first attempts to define a set of science opportunities for each of the case studies, including definition of the science complement for each. Specifically: (1) Phobos would permit the first detailed geological exploration of a small body. (2) Mars expedition provides opportunity to explore Phobos and Deimos. (3) Lunar observatory would provide potential for making major scientific advances through a series of observatories and provide opportunity for geological exploration of new lunar regions.

TABLE 4-1.- Continued

Study/report	Affected Case Study	Objective	Results
4.10 (cont.)			(4) The evolutionary case study provides opportunities for life sciences research in fractional gravity and for installing observatory elements during the lunar phase. The Mars phase will be similar to the Mars expedition with emphasis on developing long staytime capability.
4.11 Power Technology Workshop	1, 2, 3, 4	Establish a forum for power technologistics to understand and provide inputs to the OEXP case study requirements.	(1) Power system domains associated with power level and mission duration were updated; these domains include primary electrochemical storage, solar-based with regenerative storage, isotope power, SP-100 nuclear power, and advanced solar/nuclear power. (2) A preliminary figure-of-merit list for power systems was developed as a gross guide and not as a design tool.
4.12 Lunar Helium-3 Workshop Results	3	Provide information for assessing the feasibility, practicality, and advantage of mining He-3 from the lunar regolith to provide fusion power on Earth.	(1) Mining, beneficiation, separation, and return to Earth of He-3 is possible, requiring large-scale infrastructure and technology improvements. (2) Lunar oxygen production could serve as a technology demonstration. (3) He-3 may offer advantages for advancing terrestrial fusion power technology. (4) Recommendations were made for NASA to consider He-3 as an exploration program option, and for a joint NASA/DOE activity.

TABLE 4-1.- Concluded

Study/report	Affected Case Study	Objective	Results
4.13 Advanced Space Propulsion Workshop	1, 2, 3, 4	Establish a forum for reviewing, assessing, and providing inputs to propulsion system & related studies.	(1) Trajectory analyses for the purpose of the present case study were adequate, but the analysis techniques and tools will need to include more realism and higher fidelity for further work. (2) There is a concern about the assumption of applicability of aerobrake results derived from return from geosynchronous orbit. (3) Nuclear thermal could be a strong competitor for lunar and Mars missions, particularly if aerobrakes are not as capable as has been assumed.
4.14 Robotics Technology Workshop	1, 2, 3, 4	Assess mission needs, current technology adequacy, and feasibility of needed advancement.	(1) The in-space assembly process is possible with reasonable extensions of current technology. (2) Make all manipulated parts robot-friendly and seek compability between robot & EVA-friendly design.
4.15 Minimum Crew Size for Phobos and Mars Missions	1, 2	Define the primary issues that will determine the number of crew required for a Phobos or Mars mission.	Lowering the Phobos expedition to two crew members results in potentially significant degradation of mission objectives at significantly higher risk in order to gain an estimated 25% reduction in mass to LEO. The cost savings associated with mass may be countered by cost associated with increased reliance on new technology and higher reliability systems.

4.1 EARTH-MOON NODE LOCATION

The objective of this trade study is to establish requirements and provide rationale for the location of the assembly, staging, and/or otherwise aggregating function of the mission support elements for the four OEXP human exploration case studies.

All four of the case studies investigated by OEXP in FY 1988 require the delivery of large quantities of mass to Earth orbit. The choice of location for (or need of) a transportation node in Earth-Moon space to support and/or enhance these missions will depend on a number of factors. The use of a transportation node, its location, and its assigned functions may have a significant impact on overall mission performance. For FY 1988 the node location study was limited to Case Studies 3 and 4.

4.1.1 Case Study 1 Node Location

The baseline Phobos mission (Case Study 1), by design stipulation, does not require a transportation node in low-Earth orbit (LEO). It is specifically directed to not impact the then-existing Space Station Freedom, except in the area of life sciences research, and, further, to minimize Earth-orbit assembly operations. Therefore, if any LEO infrastructure is required, it is currently assumed to be the then-existing Space Station Freedom, and no trade study is required in FY 1988 to establish a baseline node location. However, future work will require selection of an assembly orbit in the event a node is not used.

4.1.2 Case Study 2 Node Location

Case Study 2 is baselined as a set of three piloted missions to Mars, launching in three consecutive Earth-Mars opportunities. Since there is no use of lunar resources, and the objective is to not develop a sustained presence on Mars, it was assumed for the current study activity that there will be no commitment of resources to build a transportation node separate from the then-existing Space Station Freedom. Hence, the node is the station and no trade study was initiated in FY 1988 to establish the baseline location. However, the orbital node and inclination of the station orbit need to be reviewed in the future for compatibility with the sprint departure and return conditions. If further study determines that a transportation node separate from the station is required, it is likely, given the expeditionary nature of this case study, that this node will be located in LEO, perhaps coorbiting with the station. This and other issues, such as the utility of a node in near-Mars space, will be addressed in subsequent studies. At this time it is felt that the expeditionary nature of this study will argue against the need for a Mars node.

4.1.3 Case Study 3 Node Location

The objective of this study is to define the preferred location, LEO or low lunar orbit (LLO), for the transportation node for Case Study 3 (Lunar Observatory), as called for in the OEXP Study Requirements Document (SRD).

The key assumption made in this trade is that the lunar vehicles will not be launched to LEO fully assembled and fueled. Because of the nearly trivial nature of this trade for Case Study 3, the study was limited to a brief qualitative consideration of the potential issues involved. No documentation has been produced for this task as it relates to Case Study 3.

Findings. If it is not assumed a priori that a node will be used in Case Study 3, four alternatives are possible: no node, LEO node only, LLO node only, and nodes in both LEO and LLO.

If no node is used, the entire 125-t vehicle must be launched from Earth as a unit, fully assembled and fueled. A launch vehicle of this capacity is not contemplated for the timeframe under consideration. Present plans are to launch all the elements at once, but not fully fueled and not assembled in the flight configuration. Since Earth-to-orbit (ETO) launch loads are greater than those experienced during the mission, an all-up, fully fueled launch also adversely affects structural design. In view of these considerations, the case without a node is considered unattractive for the baseline in which all vehicles are expendable.

If any reusable vehicles are employed, as current studies indicate may be desirable for this case study, a LEO node becomes mandatory for storage and maintenance between missions. Thus a LEO node is assumed to be included in this case study. Because of its location, an LLO node cannot serve as a substitute for a LEO node. The only issue is whether an LLO node should be included in the program in addition to a LEO node.

LLO Node Applications. Since this case study does not include utilization of lunar resources such as propellant, the principal application for an LLO node does not apply. An LLO node can have other applications, however. It could provide utilities and other support services to the lunar transfer vehicle (LTV) while the crew is on the lunar surface. However, most missions after the initial observatory setup will be for exploration at various locations. The node would therefore have to be in a high-inclination orbit to provide adequate site coverage for these missions.

In a high-inclination orbit, an LLO node could also serve as a platform for lunar monitoring, mapping, and similar tasks that would otherwise be carried out by dedicated lunar orbiters.

LLO Node Constraints. The high-inclination orbit that is necessary for the node to be of real use constrains departure from equatorial sites on the surface to the planned date 14 days after lunar arrival. Without the node, observatory maintenance missions would have an early departure capability in case of habitation module problems. Missions to nonequatorial sites would be constrained by the LTV orbit with or without a node.

Utilization of a node in a nonequatorial LLO would place an additional and unnecessary constraint on mission launch opportunities. The combination of a 14-day daylight operations requirement, which constrains launch opportunities to once a month, and the need to launch in plane, or nearly so, with the LLO node as well as the LEO node would force extended delays if a launch opportunity were missed.

Other Considerations. The LLO node would be an additional program element that entails added cost, integrations and operational complexity; it would also require an additional rendezvous and docking operation upon arrival in lunar orbit. The LLO node serves no function at all for the two cargo missions.

Conclusions. It is concluded for Case Study 3 that a LEO node is highly desirable, if not essential, and that there is no plausible reason for an LLO node. No open items have been identified that would qualify or affect these findings. No FY 1989 activity is required or planned for Case Study 3 on this trade.

4.1.4 Case Study 4 Node Location

This case study envisions the construction of a permanently manned lunar base, followed by an evolutionary branching to a permanent outpost at Mars. The intent of this case study is to push technology development; therefore, it is envisioned that there will be significant use of lunar resources in performing the Mars portion of the mission. The case study employs reusable vehicles where appropriate and may involve significant vehicle traffic in Earth-Moon space in support of the evolving Mars outpost. It is the objective of this study to determine the most desirable location in Earth-Moon space to support the transportation needs of the Mars portion of this case study.

Key Assumptions. Key assumptions used in this study are listed below:

a. The Space Station Freedom is operational in LEO regardless of the node location, and is the initiation point of all operations.

b. Mars crews return via aerocapture to the station.

c. All vehicles utilize aerocapture for returning to LEO from Earth-Moon space.

d. All chemical propulsion systems have an Isp of 485 and a propulsion system inert mass equal to 15 percent of the propellant loading.

e. Space vehicles are utilized as shown in table 4.1.4-I.

TABLE 4.1.4-I.- SPACE VEHICLE TRANSPORTATION FUNCTIONS

CHEMICAL PROPULSION VEHICLE TRANPORTATION FUNCTION

To From	Space station	Node	Lunar surface	Mars
Space station	—	Deliver crew	—	—
Node	Return empty crew module	—	Return empty LLOX tankers	Crew departure
Lunar surface	—	Deliver mission LLOX	—	—
Mars	Crew return	—	—	—

NEP VEHICLE TRANSPORTATION FUNCTION

To From	Space station	Node	Lunar surface	Mars
Space station	—	Deliver inter-planetary transport vehicle	Deliver LH2 for LLOX tankers	Deliver cargo vehicle
Node	—	—	—	—
Lunar orbit	Return for next sortie	—	—	—
Mars	—	—	—	—

f. Lunar liquid oxygen (LLOX) can be produced and used for fueling the Mars piloted vehicle and the tankers transporting the mission liquid oxygen (LOX) to the transportation node.

Approach. The study began with the premise that a transportation node has been justified by specifics of the case study and that the scope of this activity is to analyze the location of this node.

Five candidate node locations that are broadly representative of the major location options in Earth-Moon space were chosen for analysis in FY 1988 (table 4.1.4-II). They are LEO, geosynchronous Earth orbit (GEO), Earth-Moon libration point, L1; LLO; and elliptical orbit with perigee at Earth and apogee at the Moon.

To ensure the proper level of analysis, to establish a level of confidence in the results, and to understand the domain that the results are valid, it was necessary to identify the major set of considerations that potentially have a bearing on the results and should ultimately be taken into account. It was judged equally important to ensure that the identified considerations are evaluated against appropriate criteria (figure of merit). Several evaluation criteria candidates to be integrated into an overall node location assessment include minimization of initial mass to LEO, minimiza-tion of propellant used, maximization of node operations simplicity, and/or minimization of the dollar cost.

For FY 1988, the study activity identified the candidate Earth-Moon trade space and began providing preliminary analysis and observations thereof.

Findings. The preliminary overall Earth-Moon node location trade space is shown in table 4.1.4-III. Indicated in this table is the FY 1988 study emphasis on LEO mass and LLOX benefits derived from the leverage of node location on mission- and mission-support delta V.

Table 4.1.4-IV shows the delta V required for each node location, from which LEO and LLOX mass were derived. Two key points need to be emphasized:

a. Although the determination of the total delta V is instructive, it is in no way an appropriate measure of suitability or optimality of a particular location for the transportation node. It is, however, a key input to the determination of the total vehicle mass requirement in LEO and total utilization of LLOX propellant.

b. Results of this type of analysis are very dependent on the initial assumptions. A case in point is that the current analysis is based on the most favorable Earth-Moon orbital mechanics geometry that can be obtained, as well as an assumed C3 for the mission. In reality, the initial conditions established for the Earth-Moon node, the effect of gravitational perturbations, and the mission date will all tend to affect the actual mass performance, launch frequency, and quite possibly the behavior of these parameters relative to Earth-Moon node location. Since these results assume a best-case scenario, care must be taken to not oversimplify the problem and caution must be exercised in arriving at conclusions.

The results of the analysis are shown in figure 4.1.4-1. Displayed for each candidate node location are graphs depicting the initial mass to LEO (IMLEO), the amount of LLOX used for the Mars mission, and the amount of LLOX used in transporting the Mars mission LLOX to the transportation node. Figure 4.1.4-1 intentionally does not show absolute performance values. Because of the caveats and concerns previously stated, because of concerns that the simulation and analysis requires incorporation of a higher level of sophistication and operational reality, and because this study is just one subset of the entire trade space, it is premature to draw any conclusions or even infer that conclusions can be drawn. At best, these data represent only a trend for a specific set of assumptions.

Within the limitations implied by the validity of the specific set of assumptions used, the following observations were made:

a. The location of the node in near-lunar space (L1, LLO, Cycler) appears to be a better choice for steady-state operations if LLOX is used; these locations (theoretically) result in improvement in LEO mass requirements, mission LOX requirements, and LOX transport requirements over near-Earth space.

b. Any further study concerning the utility of a node in LEO must address the issue of LLOX versus Earth-based LOX use. As shown in Figure 4.1.4-1, the use of LLOX appears to significantly reduce the Mars-to-LEO requirements. However, the use of LLOX significantly increases the amount of LLOX required due to the transportation of the mission LOX from the Moon to LEO. Also, a previous study by Eagle Engineering in 1985 concluded that it will be difficult to deliver LLOX to LEO at less cost than to deliver LOX from the Earth surface. This point signals the obvious link between this trade study and the Extra-terrestrial Propellant Leveraging Trade Study, a link that requires further investigation.

Issues/Open Items. One of the questions that arises from this study is the need for a transportation node to support the lunar phase of the mission before departing for Mars. Assessments of Case Study 3 requirements have indicated

TABLE 4.1.4-II.- CANDIDATE NODE LOCATIONS

LEO	500-km altitude circular orbit about Earth
GEO	24-hr circular orbit; 42,242-km radius
L1	Earth-Moon L1 point; 322,000-km radius
Cycler	Elliptical orbit in Earth-Moon space; 6878-km perigee x 384,000-km apogee
LO	110-km altitude circular orbit about Moon

TABLE 4.1.4-IV.- DELTA V REQUIREMENTS (m/sec)

		Transport crew to node (1)	Transport vehicles to node (1)	Transport LLOX to node (2)	TMI (@ C3-30)	Total
Node Location	LEO	0	0	8290	4470	12760
	GEO	5450	5450	7080	3540	21500
	L1	4440	4440	5020	2050	15950
	Cycler	3380	3380	5100	14210	13270
	LO	4820	4820	3480	2230	15330

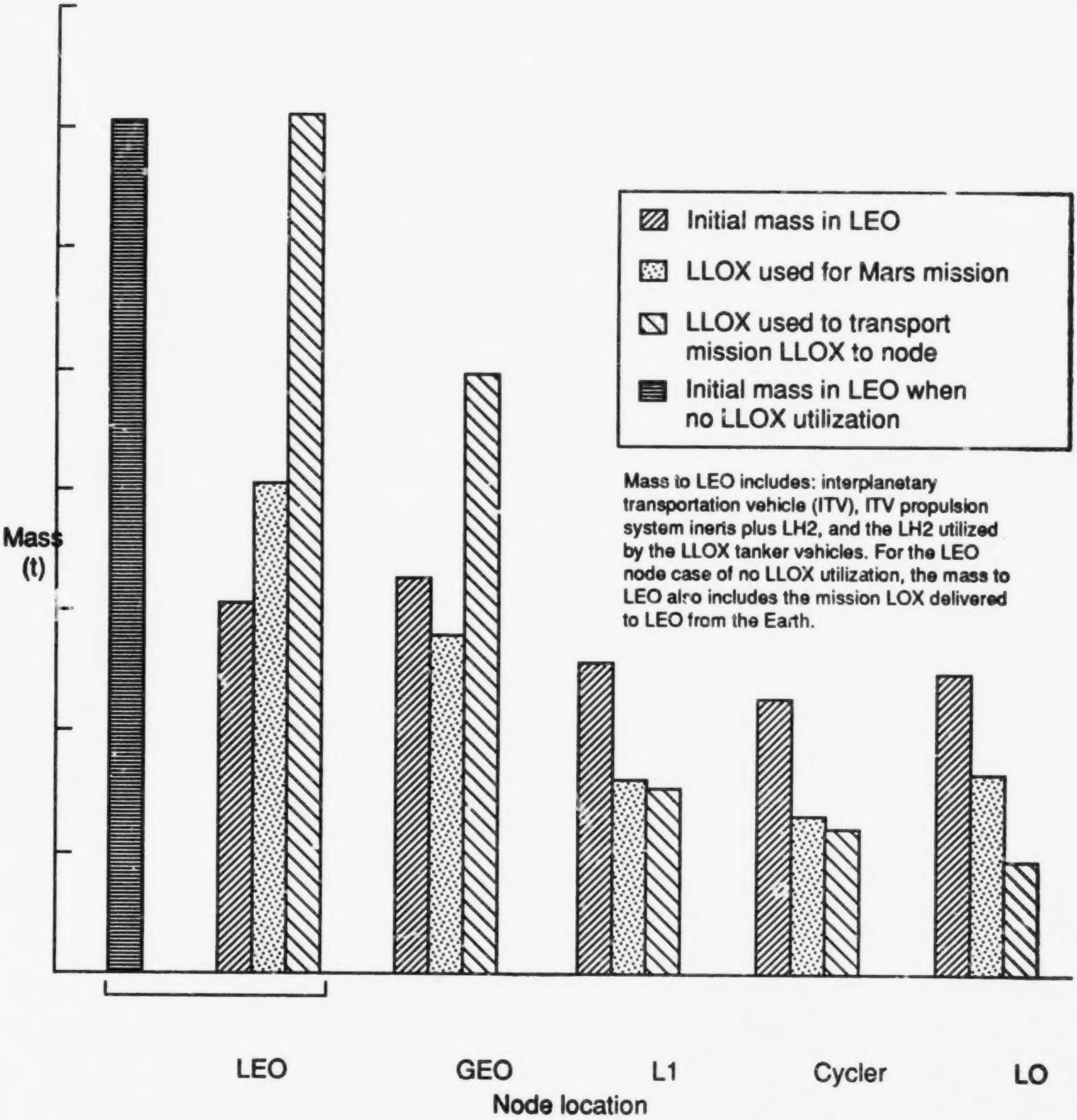
(1) Round-trip: To/from LEO; with aerocapture at Earth

(2) Round-trip: To/from lunar surface

TABLE 4.1.4-III.- EARTH-MOON NODE LOCATION TRADE STUDY

Considerations \ Candidate optimization criteria	Initial mass to LEO	LLOX used	Operations simplicity	$ Cost
Node infrastructure - Transport - Assembly - Maintenance operations				
Node crew support - Rotation frequency - Assured crew return - Consumables - Health				
Vehicle logistics - Assembly checkout - Maintenance operations - Turnaround requirements - Fleet size				
Orbital mechanics effects on departure return - Launch windows Duration and frequency - Inclination effects - Node and plane misalignments				
Launch vehicle performance - ETO transportation - Injection stages				
Mission crew return aspects - Frequency - Return location Direct entry to earth Return to space station Return to transportation node	X	X		
Extraterrestrial propellant production - Phobos - LLOX	X	X		
Performance aspects - Transport mission crew to node - Transport propellant to node - Transport vehicles to node - Trans-Mars insertion	X X X X	X X X X		

X - Analysis performed as part of FY '88 study.



Mass to LEO includes: interplanetary transportation vehicle (ITV), ITV propulsion system ineris plus LH2, and the LH2 utilized by the LLOX tanker vehicles. For the LEO node case of no LLOX utilization, the mass to LEO also includes the mission LOX delivered to LEO from the Earth.

Figure 4.1.4-1.- Node location mass performance results.

that a LEO node is highly desirable if not mandatory and that a lunar orbit node is not needed (section 4.1.3). An assessment of Case Study 4 lunar phase requirements is required to determine the applicability of Case Study 3 conclusions. Should the findings be similar, a transition from a LEO node to the Mars departure node would have to be incorporated.

The current FY 1988 results were based on performance considerations only. The analysis in effect assumed that the LLOX plants on the Moon were already in operation, and the construction and setup costs have been charged to a previous phase of the case study; however, no existing production capacity was considered. Continuation of this study, broadened to be more inclusive and detailed, is needed.

Planned FY 1989 Studies. This study should continue into FY 1989, updated as required, to incorporate changes to the Case Study 4 requirements.

4.2 EXTRATERRESTRIAL PROPELLANT LEVERAGING

The objective of these studies was to assess the impact and potential benefit of augmenting Earth-supplied resources with extraterrestrial (lunar, Phobos, Deimos, or Mars) resources.

Background. A number of independent conceptual studies have been done since the early 1970's on the potential of using extraterrestrial materials to augment the Earth materials needed for space development, thereby lowering transportation costs from the surface of the Earth to low Earth orbit (LEO). Once out of Earth's gravity field, it is relatively inexpensive to transport large masses in space. Even the prospect of lifting materials out of the Moon's gravity field at one-sixth of Earth's gravity, or from Mars at one-third of Earth's gravity, may be viable options.

Many materials are available from the lunar, Phobos, and Deimos regoliths, as well as the Mars regolith and atmosphere. The primary focus in the past has been on potential propellants, principally oxygen and hydrogen since there is an obvious market in space for these resources. Other materials may also offer significant potential, depending on the specific scenario. These potential resources include oxygen, silicon, iron, calcium, aluminum, and trace amounts of hydrogen, carbon, and nitrogen from the Moon; oxygen, hydrogen, and carbon compounds from Phobos or Deimos; and oxygen, hydrogen, iron, carbon, and argon from Mars. Another potential class of resources, extremely rare or nonexistent on Earth, would actually have markets on Earth. The first example of this class, found in the lunar regolith, was helium-3, which could be used as a fuel in fusion energy generation. Other unknown resources may exist. One final possibility is the use of bulk regolith as radiation and micrometeorite shielding, both on planetary surfaces and in space.

Demonstrating the viability of extraterrestrial propellant (ETP) is exceedingly complex because it involves a large number of interdisciplinary and tightly interrelated variables. To determine these variables it is necessary to understand space development options to a level of engineering and programmatic detail that does not currently exist. These variables also tend to change dramatically, depending on the particular scenario; therefore, the viability of ETP use and the associated implementation plan must be demonstrated for each scenario.

An additional concern is that some analyses suggest increased program costs with lunar oxygen use. However, within the framework of long-term resource production plans and space development in general, ETP use may still be viable. For example, it is unlikely that economic viability of human settlements is possible without extraterrestrial resource use to some degree. Early ETP capability development with higher initial capital costs may be justified by its contribution to expertise in later broad-based resource utilization.

Brief descriptions of recent JSC-sponsored studies are given in the **Products** paragraph of section 4.2.1. As stated, the results of these studies are strongly driven by the development scenarios that each considered. The most difficult parameters to determine are the capital and operating costs of the production facilities. It may be possible to minimize costs, particularly operating ones through automation and robotics; however, many of the studies neglected costs and considered only whether propellant production was feasible. Generally, the results were positive for the development scenarios considered. Since it is not yet possible to reliably compute actual costs, masses are used as cost indicators for the studies where cost is considered. This relationship is only nominally acceptable, since many other factors will affect the overall cost of space development and operations.

4.2.1 JSC-Sponsored ETP Studies

The advantages of ETP are highly scenario-dependent. They are based on

a. The market in space for ETP's

b. Earth-to-orbit (ETO) transportation costs

c. LEO/low-lunar-orbit (LLO) and LLO/lunar-surface (LS) transportation costs

d. Capital and operating costs of the ETP production facilities

Key Assumptions. The JSC in-house scenario considers facility masses, but assumes very aggressive lunar development, with emphasis on science, resource production, and habitation (in that order).

The JSC Eagle Engineering study considers a generic evolutionary lunar/Mars scenario, and includes rough calculations of the production facilities mass. This study also assumes that two manned missions are required before propellant production is achieved.

Approach. The JSC in-house study ran two cases of the LSPI-lunar base model, the first using lunar liquid oxygen (LLOX) production and the associated facilities, with LLOX used round-trip from LLO to LS; the second without LLOX production facilities, importing all propellants from Earth. The lunar base model was not designed to consider LLOX usage, so the propellant import requirements were determined from a hand calculation using delta-V/mass-fraction equations.

The JSC Eagle study used a series of in-house models developed to consider lunar, Phobos, and Mars propellant production and the associated trajectories and market locations for a manned Mars mission.

Products.

Previous Studies.

a. "Design of a Lunar Colony," 1972 - First characterization of a lunar base with a lunar oxygen production facility. Does not include the economic or transportation impacts of LLOX export.

b. "Lunar Surface Return Study" (JSC in-house), March 1984 - Considers two options for LLOX usage round-trip LS/LLO and round-trip LLO/LEO.

c. "Analysis of Lunar Propellant Production" (Eagle Engineering), December 1985 - Primarily an economic-impact study of producing both LLOX and lunar liquid hydrogen (LLH_2), with main consideration the delivery of LLOX to LEO.

Related Studies.

a. "Lunar Surface Operations Study" (Eagle Engineering), December 1987 - Performed to develop a mission manifest for a selected lunar base scenario, determine the nature of surface operations in the scenario, propose concepts for using machines/remote operations to perform repetitious or hazardous surface tasks, and present a preliminary crew extravehicular activity/intravehicular activity (EVA/IVA) time/resource schedule for conducting the missions.

b. "Conceptual Design of a Lunar Oxygen Pilot Plant" (Eagle Engineering), July 1988 - A study to provide a list of candidate lunar oxygen production processes, develop a rationale for selecting two processes for further study, produce conceptual designs of pilot plants based on the two leading processes, determine impacts of pilot and production plants on base operations, and determine the feasibility of recovering solar-wind-implanted hydrogen from the lunar regolith.

FY 1988 Propellant Leveraging Products.

a. LLOX Production and Facilities Impact Study - JSC in-house

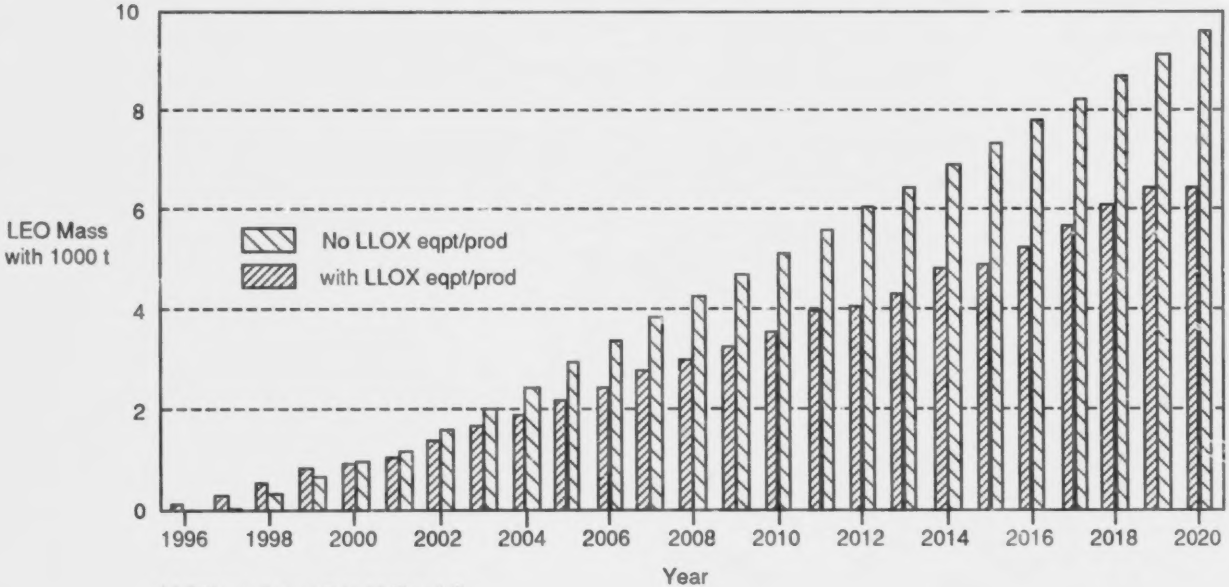
b. Extraterrestrial Propellant Production - Eagle Engineering

Findings.

JSC in-house. Considering preliminary facility and operating masses, LLOX use for the round-trip LLO/LS indicates increased costs in the short term, but indicates a savings of about 30 percent over the long term (20 years), as shown in figure 4.2.1-1.

Eagle Engineering. The impact of propellant production on a manned Mars mission was investigated. Using rough calculations on facility masses, Mars surface propellant production could result in 5 percent reduction in LEO mass for LOX only and 7 percent for LOX and fuel; Phobos propellant production could result in 15-20 percent savings for LOX and 25 percent for LOX and fuel. If propellants were produced on both Phobos and Mars surface, LOX production alone could result in 30 percent reduction, while producing both LOX and fuel could reduce LEO mass by 44 percent. These LEO mass savings are shown in table 4.2.1-I, the propellant production options in figure 4.2.1-2. Break-even points and potential long-term reduction in LEO mass are given in figure 4.2.1-3. The absolute break-even point in time is very sensitive to the assumption of first usage; but it has been shown to occur by the second usage in most cases.

There may be other considerations (depending on the scenario) for propellant production other than LEO mass or near-term program costs, including self-sufficiency, flexibility and safety, and technology development for other products needed for human settlements. ETP production is beneficial only for long-term development; sprint missions do not benefit. Facilities startup can take a long time to achieve full production capability, and



LLOX used roundtrip LLO to LS

Figure 4.2.1-1.- Lunar base development: Accumulated LEO mass required for LLOX production vs. no LLOX.

TABLE 4.2.1-I.- MANNED MARS MISSION LEO MASS SAVINGS FROM PROPELLANT PRODUCTION

Results	
• Lunar oxygen in lander	- Saves up to 30% of LEO mass
• Mars surface propellant	- O_2 only saves 5% - O_2 and fuel save 7%
• Phobos prop. production	- O_2 only saves 15-20% - O_2 and fuel save 25%
• Phobos and surface	- O_2 only saves 30% - O_2 and fuel save 44%
• Lunar oxygen for Mars missions	- Need hydrogen from Moon or other fuel

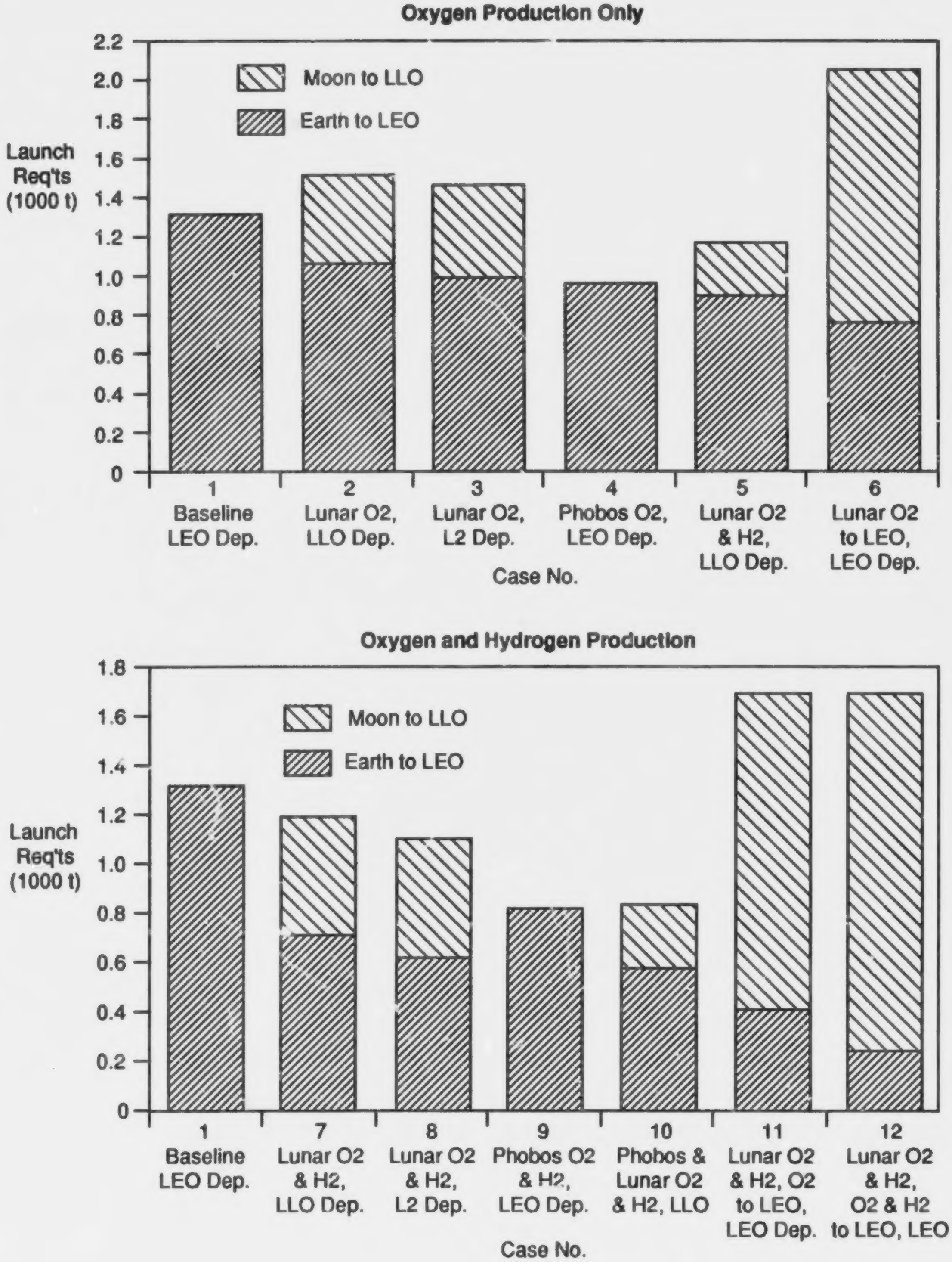


Figure 4.2.1-2.- Manned Mars mission launch requirements versus propellant product options.

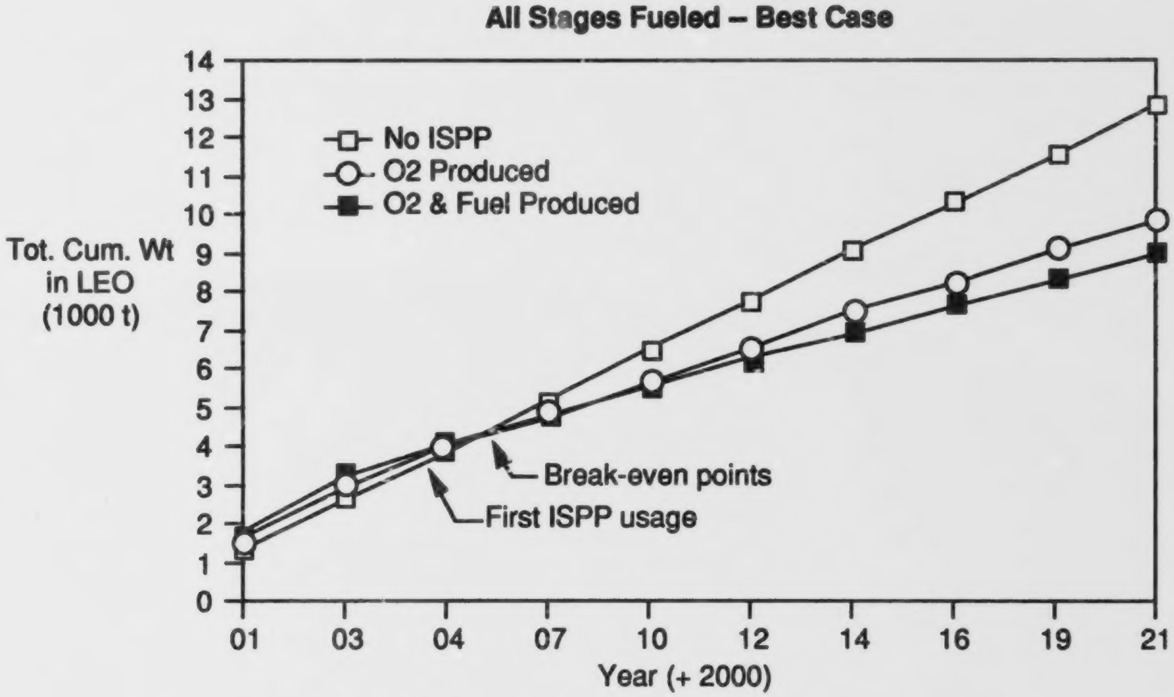


Figure 4.2.1-3.- Manned Mars mission Phobos & surface insitu propellant production.

payback is realized only over a 10- to 20-year time scale.

Issues/Open Items. The additional mass required for production facilities construction and operations is still only poorly defined, if at all, particularly for Phobos and Mars facilities. In addition, the contribution of the masses and other factors to the overall cost is very poorly understood.

The tangible benefit of establishing ETP production facilities is uncertain. Hindsight may show that the major benefit of these facilities is not their economic savings, but the technologies and capabilities spinoffs for both space and Earth applications. It is not clear how to factor these considerations into the analysis.

Planned FY 1989 Activity. Individual pieces of the puzzle have been assembled. An integrated study needs to be done to apply the ETP production analysis, including that for facility masses, to the OEXP case studies. An understanding of the contribution of the masses and other factors to the overall program costs is also needed. Finally, the benefits of generating less understood products and of getting started early on ETP production need to be defined.

4.2.2 In Situ Propellant Leverage Analysis

The objective of this study is to determine the effects of using lunar, Phobos/Deimos, and Mars propellants in lunar, Mars, and lunar/Mars blended scenarios.

Background. Current concepts for lunar/Mars blended scenarios involve the production of lunar oxygen and in situ propellant production on Phobos and Deimos. It is important to ascertain which space resource production and use scenarios have the most performance, cost, and strategic benefits.

Key Assumptions. The key assumptions of this study include:

a. Insitu propellants are transported by cryogenic tanker vehicles similar to the OTV designed by General Dynamics Space Systems (GDSS) Division (Isp = 485 sec; insulation and meteoroid protection require 10 percent increase in inert tank weight scaling relations; zero boiloff losses assumed).

b. Aerobrakes are scaled to 15 percent of entry weight.

c. Payload weights are from OEXP Study Requirements Document (SRD) (Lunar Outpost to Early Mars Evolution case study).

d. Both LH_2 and LO_2 are produced at Phobos/Deimos.

e. Conjunction class trajectories are used.

f. Infrastructure buildup is not included in initial cases.

Approach. Analex Corporation and GDSS analyzed the leverage associated with in situ propellant production. GDSS computer models relating sites and products, delivery trajectories, users, and economics are used to obtain quantitative insights into space resource utilization for the Moon and Mars. Space Operations Analysis Resource (SOAR) is an interactive, user-friendly computer program (including a library of orbital mechanics routines) used to perform multivehicle mission planning. SOAR, initially an Earth-centered simulation, has been expanded to Moon- and Mars-centered and interplanetary trajectories (Sun-centered). Space Transportation and Resources (STAR) is a user-friendly spreadsheet model that simulates personnel and cargo transportation between an arbitrary number of space transportation nodes. STAR models refueling, staging, and/or payload changes at any node. Vehicle scaling (including aerobrake) relations are entered into STAR, as are trajectory data (e.g. delta V's from SOAR). A Mars-lunar transportation/resource cost model is being expanded and fully integrated into STAR.

Findings. The myriad of possibilities is best understood by considering each propellant application site separately, then comparing the production needed to deliver a fixed amount of propellant payload to that usage site (or destination). Figure 4.2.2-1a through -1d show performance data grouped by payload destination. For all locations the payload delivered is 100 t. Aerobraking is used whenever possible and all vehicles are reusable. Short bars indicate efficient transport situations with relatively high payload fractions. The bar segments show how the propellant is consumed.

Figure 4.2.2-1a shows LEO as the destination. The Earth mass driver is shown as a reference case since no propellant is expended. Propellant transport from Phobos/Deimos (Ph/D) is most promising, with 95 t of propellant used to transport 100 t to LEO. If lunar hydrogen and LLOX are available, their transport would also be favorable (215 t used). However, if LLOX is available but H_2 must be transported from Earth, propellant used in the LTV and LL is increased because hydrogen must be transported to the Moon as well as from the Moon. The major inefficiency of this method results from transporting hydrogen to LEO from Earth's surface (1106 t, bringing total propellant use to 1409 t). Propellant transport from the surface of Mars requires 1037 t. The most demanding transport case is launching propellant from Earth's surface (1926 t to place 100 t in LEO).

Figure 4.2.2-1b shows the LLO destination. Aside from a lunar mass driver (reference case), lunar propellants (LLOX and LLH_2) are most promising, requiring 60 t of propellant compared to 1133 t if hydrogen must be transported from Earth. Propellant transport from Ph/D is very favorable, requiring 187 t. Transport of Mars surface propellants requires 1292 t. Again, highest energy cost is transport from Earth's surface (2089 t).

Figure 4.2.2-1c shows Mars orbit as the destination. Production directly in Mars orbit, on Phobos or Deimos, is the reference case. Propellant transport from Mars surface requires 294 t. The LLOX/Earth H_2 case is relatively complex, transporting hydrogen from Earth and LOX from the Moon to drive all vehicles, with LLO serving as the staging point; this operation uses 1608 t, or 1264 t for the expendable launch vehicle (ELV). If all propellant comes from Earth's surface, 8588 t are used for transport (8280 t for the ELV).

Figure 4.2.2-1d shows the lunar surface destination. LLOX and Ph/D hydrogen is the favorable prospect (296 t used). Here, LLOX and Ph/D hydrogen are used to drive the LL. Ph/D propellant is used in the PTL. Ph/D propellant use for both vehicles is 373 t. LLOX/Earth H_2 transport requires 1568 t (1245 t for the ELV). If all propellant comes from Earth's surface, 10,854 t are used (10,440 t for the ELV).

Table 4.2.2-I shows propellant production and use locations for several possible scenarios including the seven lunar and Phobos/Deimos cases investigated at this time. The ideal mass savings (IMS) ratio figure of merit is a measure of the LEO vehicle weight savings for a given amount of refueling at the place of application (e.g. LLO) versus a scenario in which all propellants originate in LEO. As in all cases in this report, data are for steady-state scenarios and do not reflect space infrastructure buildups.

The largest IMS (4.3) is for the use of Phobos/Deimos propellants in Mars orbit. The negative IMS for lunar oxygen export to LEO suggests it will be difficult for chemical propulsion systems to make this case profitable. Using lunar oxygen in LLO (IMS = 1.8) appears to be a very good case, although it has lower leverage than using Phobos/Deimos propellants near Mars. The very good leverage for exporting lunar oxygen to Mars orbit is reflective of the fact that only one of the three vehicles required in case 7—the lunar tanker which delivers hydrogen to LLO—ever appears in LEO. The extremely favorable leverage of Phobos/Deimos propellants in LMO and the surprisingly good performance in LEO and LLO

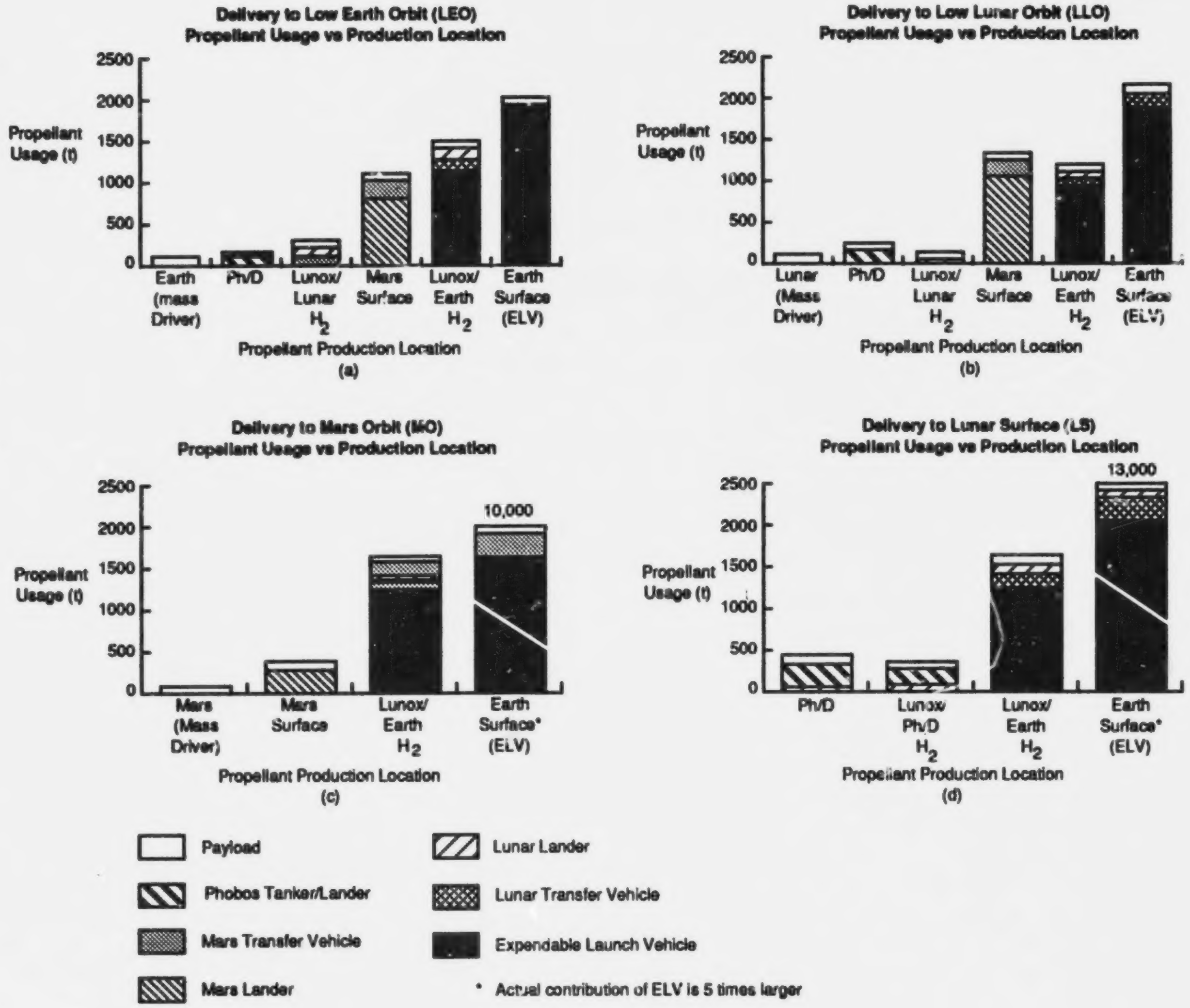


Figure 4.2.2-1a - d.- In situ propellant leverage performance data.

TABLE 4.2.2-I.- ISP PROPELLANT LEVERAGE ANALYSIS RESULTS - STATUS

Place of Application	Ph/D $H_2 + O_2$	LLOX/2	Mars O_2 + Fuel
LEO	#3 IMS =.4	#6 IMS =-.6	TBD
Lunar surface	TBD	TBD	TBD
LLO	#4 IMS =.4	#5 IMS =1.8	TBD
E-M Lib	TBD	TBD	TBD
Mars surface	TBD	TBD	TBD
Mars orbit	#1, #2 IMS =4.3	#7 IMS =1.8	TBD
E-S Lib	TBD	TBD	TBD

Figure of merit: IMS ratio: t of LEO savings per t refueling. Above results based on conjunction class LH/LOX freighters w/aerobrakes suggest strongly that in situ propellant production on Phobos/Deimos may be the most profitable early space resource scenario.

Conclusions. Our current conclusions and recommendations are summarized in table 4.2.2-II. Phobos/Deimos exploration and propellant production is an attractive prospect that may support operations near Mars, the Moon, and LEO, while providing an economic incentive to explore Mars. Lunar oxygen can be beneficially used in the lunar vicinity.

4.3 THE FEASIBILITY OF AUTOMATING LUNAR LOX PRODUCTION

Study Overview. As humans spend longer periods of time on the surfaces of the Moon and Mars, the use of in situ resources will become more and more attractive. The

TABLE 4.2.2-II.- IN SITU PROPELLANT CONCLUSIONS AND RECOMMENDATIONS

Where should the priority be for leveraging chemical propulsion systems to Mars? On LLOX for outbound savings? On Ph/D propellants for homeward bound savings?	Should in situ propellants be exported to LEO?
Conclusions	Conclusions

• Ph/D propellants offer substantially more LEO mass savings and should be exploited with top priority.
• More leverage from Ph/D because:
  • Ph/D is regularly "closer" to LEO than is the lunar surface, via conjunction class chemical freighters
  • Ph/D potentially offers easily available fuels, as well as oxidizers.
	• Ph/D propellant might be beneficially exported to LEO, given adequate excess production levels, good propellant storage, and highly automated plants.
	• Lunar oxygen cannot be beneficially exported to LEO. using chemical freighters.

In situ propellant leverage analysis conclusions

- In situ propellants offer great potential for space exploration cost savings.
- Phobos and Deimos offer the greatest potential leverage on all missions involving Mars.
  - Exploitation at the earliest possible opportunity is strongly recommended.
  - Science and engineering precursors should precede the first human visit.
  - The Ph/D leverage will be very high even if hydrogen is not found.
- Labor-intensive operations of ISPP plants will markedly degrade their advantage. Strong emphasis should be given to optimizing automation.
- Lunar oxygen can be attractive for lunar vicinity operations, return to Earth, and outbound sprints to Mars.
- Phobos/Deimos propellant production may be a low technology alternative to multi-megawatt nuclear cargo vehicles which allows an evolutionary buildup of mass through LEO in the Lunar Outpost to Early Mars Evolution scenario.

Potential high payoffs

- Ph/D fuels (hydrogen or other) with LLOX for lunar vicinity operations
- Ph/D hydrogen for nuclear thermal rockets
- Ph/D excess oxygen is ideal for Mars vicinity life support

production of lunar liquid oxygen (LLOX) is a high-leverage item because it can provide inexpensive oxygen for rocket fuel and for life support, and studies (Criswell, 1979; Heiken, 1972; Laul and Schmitt, 1973)[1] indicate that oxygen exists in sufficient abundance at the lunar surface for these needs.

OEXP Case Study 4 incorporates as baseline a lunar oxygen production plant, which will require initial setup and checkout, plant operations including mining and materials transport, and repair activities. An in-house study examined the feasibility of the operation of an automatic LLOX production facility. The study specifically examined mining and waste removal processes.

Results indicate that LLOX production can be automated by using existing automated processing technologies. Initial setup and contingency operations still need to be addressed as they may be the hard issues in the automation of a LLOX production facility.

The objective of this study is to examine the feasibility of automation in digging, acquiring, and transporting lunar soil and in the operation of a LLOX production facility. Two specific goals of the study are to prove the possibility for an automatic LLOX production site, and evaluate the feasibility of automating the specific processes being assumed in Case Study 4.

Methodology. There is much Earth-based knowledge in the field of automated mining. As early as 1960 an iron ore reclaiming facility was automated to the extent that one person was able to supervise the entire 2-square-mile plant. The feasibility of an automated LLOX plant was demonstrated by comparison with terrestrial analogs.

Key Assumptions. The Case Study 4 annual requirements for LLOX, not considering losses, equates to approximately 800 t/yr. Allowing for losses and uncertainties, a total annual requirement of 1000 t LLOX is assumed. The following analysis can be proportionately scaled for different annual requirements.

Plant Analysis. The oxygen on the Moon appears to be most often bound in the four oxides FeO, SiO_2, Al_2O_3, and TiO_2. Oxygen can be extracted by chemical, electrolytic, and pyrolytic processes (Lindstrom, 1979; Grodzka, 1977; Waldron, 1983). Two options examined in detail are the reduction of ilmenite using hydrogen as a reducing agent, and the direct electrolysis of lunar soil (Kesterke, 1971; Carroll, 1983).

Our concern here is thus restricted to the automatic acquisition in sufficient quantities of the appropriate lunar material, to its automatic delivery to the chemical plant for the production of LLOX at an average rate of 1000 t/yr, and to the automatic removal of the residue material. The total quantity of lunar material that must be acquired in raw form depends very much on its oxygen content and yield, as well as on the efficiency of the chemical decomposition and extraction process. The range of the average mining rate is between 3 and 80 t/hr. The low estimate assumes electrolysis of lunar mare basalt; the high estimate assumes hydrogenation of lunar material with the characteristics of ilmenite from the Apollo 11 site (Christianson, 1988).

Operating Requirements. The operation of a LLOX plant is highly dependent on the mass and volume of material passing through it. These numbers may vary by several factors of 10, depending on the choice of oxygen extraction technique. This mining rate variance will impact the scope of the material mining and handling operations. This impact will be reflected in the durability of the equipment used. It will also influence the complexity and reliability of implementing these operations.

Compared to operations on Earth for any given mining technique, the volumes to be dealt with are similar, whereas the total weight is one-sixth that on Earth. This weight decrease should provide relief of gravity-induced stresses (important for high volume rates), but will not reduce acceleration-induced stresses.

Typical of the oxygen extraction techniques reviewed for this study is one based on an electrolytic process suggested by Waldron (1988), with the following operating requirements:

Daylight operations (photovoltaic power source)	4000	hr/yr
LLOX production rate	250	kg/hr
Minimum soil mining rate	3000	kg/hr
Mass of mining equipment	10000	kg
Theoretical mean decomposition energy	5	kWhr/kg(O_2)
Theoretical decomposition power	1250	kW
Preheating of ore	1700	kW
LLOX liquefaction	250 (1 kWhr/kg)	kW
Mining operation energy	negligible	
Total power estimate	3200	kW

Conceptual Design. It is assumed that the lunar surface mining and oxygen production site has been selected according to appropriate material characteristics. From what is currently known, lunar soil is generally finegrained with some rocks and boulders; however, heavy digging and crushing appears unnecessary. With a mining volume of 12,000 t/yr, requirements for several years of operation can probably be mined on one lunar site. Fifteen years of operations would require a mining area of approximately 100,000 m^2, if the average mining depth is about 2 m. Figures 4.3-1 and -2 are sketches of a conceptual layout of a possible arrangement of the various elements.

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[1] Parenthetical references are to the List of Sources at the end of section 4.3.

The facilities are erected primarily by astronauts supported by automatic tools and machinery as required. The process starts with the acquisition of the lunar material by some continuously operating bucket wheel or bucket chain excavator, a kind of automated operation done similarly in strip mining on Earth. The buckets empty their contents onto a conveyor belt system which brings the material to the soil processing and oxygen production plant for further automatic processing. Before the oxygen is extracted, there may be a physical beneficiation process, depending on the particular oxygen extraction method used. In any case, the beneficiation process does not require automation techniques that are not already being used on Earth.

At most, 9 percent of the processed material is extracted and converted into LLOX. The 91 percent residue must be removed and redeposited on the lunar surface. The residue can be removed from the LLOX plant by conveyors running parallel in the opposite direction to the collection conveyors and continuously deposited by a redeposition conveyor that moves behind the bucket excavator at the same forward speed.

The entire process will proceed automatically, except for maintenance, repair, and unusual events such as an unforeseen entanglement or breakdown. The case study baselines continuous human presence on the surface. Complete automation of the plant, including attending unforeseen problems, would require substantial advancements in technology. However, Earth analogs indicate that with a high degree of automation and reliability, one person should be sufficient to maintain operations of a LLOX plant.

Findings. The required material-handling capacity of the mining and transport equipment on the lunar surface is critically dependent on the particular oxygen extraction process. It is therefore important to select a process that is efficient for material mining requirements. In this analysis, this point has been considered only from the aspect of automation and robotics, while other tradeoff considerations, such as the mass for the oxygen extraction facility and its transport from Earth to the Moon, have been neglected.

Continuous digging equipment such as bucket excavators and shiftable conveyor systems is used on Earth for mining. Except for specific adaptations to lunar conditions, such as tribology and extreme thermal changes, the required automation and robotics technology for this mining and transport equipment is the kind currently used in Earth strip-mining and ore-processing industries. It should therefore be possible to perform the mining and transport functions on the lunar surface with a minimum of human attendance, assuming a high degree of equipment reliablity and modularity of replacement parts.

Planned FY 1989 Activities. Two issues to be addressed in FY 1989 are initial setup of the production plant and contingency operations. A more general issue is the utility of automating in site preparation, including the setup of facilities for use of in situ resources.

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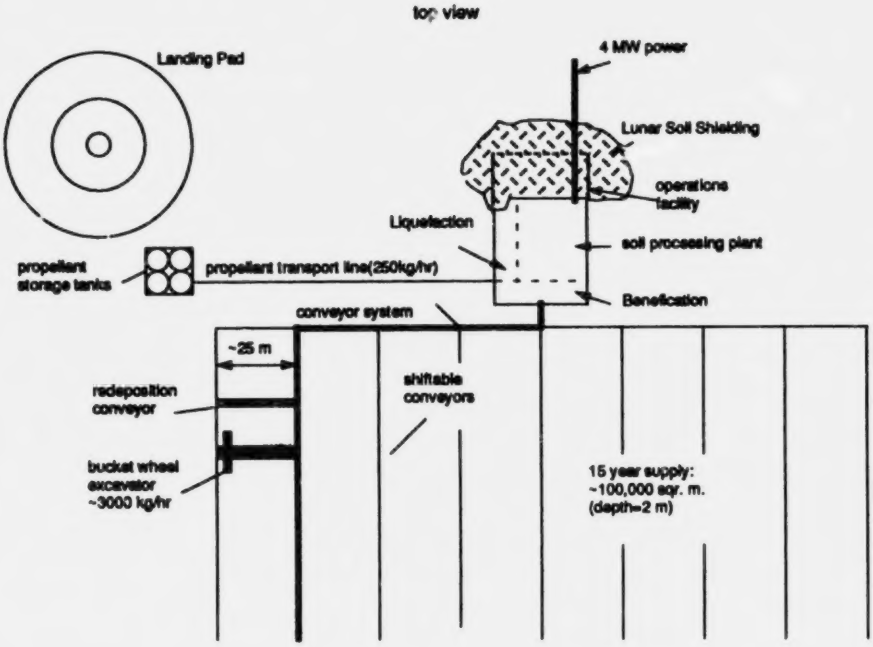

top view
Landing Pad
4 MW power
Lunar Soil Shielding
operations facility
Liquefaction
soil processing plant
propellant storage tanks
propellant transport line(250kg/hr)
Benefication
conveyor system
~25 m
redeposition conveyor
shiftable conveyors
bucket wheel excavator ~3000 kg/hr
15 year supply:
~100,000 sqr. m.
(depth~2 m)

Figure 4.3-1.- Conceptual layout–top view.

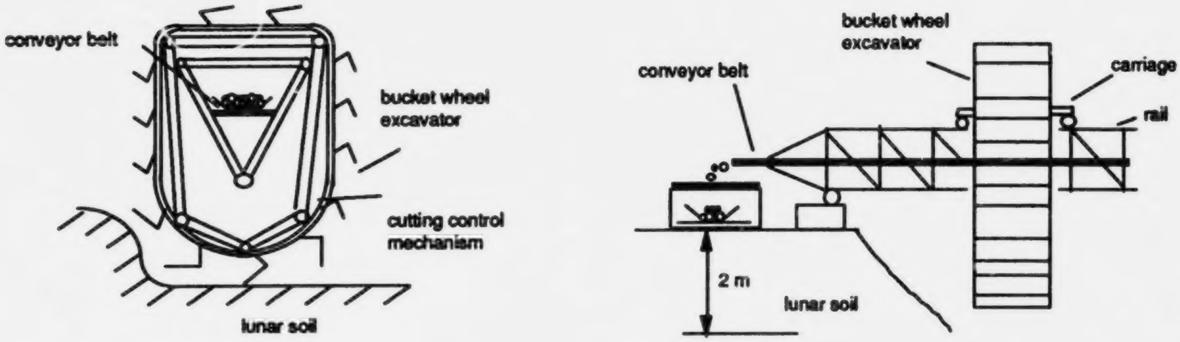

conveyor belt
bucket wheel excavator
cutting control mechanism
lunar soil
bucket wheel excavator
conveyor belt
carriage
rail
2 m
lunar soil

Figure 4.3-2.- Conceptual layout–front and side views.

4.4 LOW-EARTH-ORBIT (LEO) ASSEMBLY STRATEGY

The objective of this activity is to define and evaluate system capability and operational technique alternatives that can be employed to meet the need for a human exploration space vehicle assembled and checked out in low Earth orbit (LEO).

Background. Each of the case studies examined in this report stipulates requirements for assembly in LEO of the cargo and piloted space transfer vehicles (including their payloads). Space transfer vehicle assembly is the incremental process by which piece-part cargos delivered from Earth to LEO by unmanned and manned spacecraft are retrieved, aggregated, outfitted, checked out, and loaded with propellant and consumables in preparation for their departure for destinations beyond LEO.

For the Phobos expedition case study, the goal was that this assembly process be accomplished without the use of a special onorbit support facility, thus implying automatic and/or STS-assisted techniques; for the other case studies, the presumption was that a LEO transportation node of considerable size and capability would be required, itself providing a sophisticated stable of automated support equipment as well as a pressurized habitat for the assembly crew. Implementation of either case, however, will require an efficient and effective long-term plan which facilitates the development and integration of multiprogram requirements for space machinery, manpower, and operations capabilities (i.e., an assembly implementation strategy).

It is the purpose of this section to develop a framework about which an assembly implementation strategy can be developed as required to support each case study assessed by EXP. For FY 1988, emphasis has been in three areas: 1) the development of a functional approach to assessment of programmatic and technical assembly requirements, and an initial assessment of the Human Expedition to Phobos Case Study with respect to the requirement for no assembly node; 2) initiation of a study to determine the types of assembly-related activities which are best performed (or pre-integrated) on the ground compared to those which, with reasonable assumptions concerning level of automation and orbital operations capabilities, could best be performed onorbit; 3) rudimentary definition of LEO assembly operations support systems and techniques.

It is noteworthy that evaluation of assembly concepts specific to FY 1988 case studies requires identification of specific mission objectivities as well as specific vehicle configurations and support needs. As would be expected, much of this information arrived late in this initial study cycle, so a good deal of this year's work in the assembly area has involved the preparation of a methodical approach to programmatic and technical analysis which can be applied at the outset of the FY 1989 study cycle using FY 1988 case study vehicle configurations and needs as initialization data.

4.4.1 Development of a Functional Approach to Assembly Analysis

Like the assembly process itself, development of an approach for programmatic and technical analysis of the myriad systems studies and trades related to in-space assembly of very large spacecraft is an incremental activity. It begins with the identification of key functions within NASA's manned space program which will be affected by a major in-space assembly task. These functions are often interrelated and any strategic, programmatic end-to-end assembly plan must shape them in a manner which, though perhaps not optimum for each function as a separate entity, provides a technically achievable and cost-effective approach to the implementation of all functions as an integrated assembly "system." As specific configurations, techniques, and constraints begin to emerge, this approach identifies the technical options available within each functional area, quantifies the appropriate figure-of merit parameters for evaluation, and performs the vehicle and support system assessments and trades as necessary to verify (at a preliminary level) the feasibility of the concept and to offer suggestions on how hardware or operational modifications might result in reductions on assembly resource requirements.

4.4.1.1 End-to-End Assembly Functions

Background. Three major questions will need to be addressed as prerequisite to developing and adopting an assembly strategy for any given case study: What functions will be performed in space; how are these functions intended to be implemented; and what are the constraints that may be imposed upon or result from implementation? To answer these questions, it is necessary to identify the various parameters that may shape assembly, to obtain estimates relative to the amount of influence these parameters may have on assembly and vice versa, and to identify and perform the necessary integrated studies. The net result is that compromises to desirable assembly goals and objectives such as reduction in number of assembly flights, reduction of impact to the node, reduction of assembly sequence duration, and reduction in size of vehicles and launch packages may be required. The challenge will be to arrive at the optimum assembly solution.

Approach. For FY 1988, it was clear that an understanding of generalized assembly-related issues and/or options was necessary before arriving at any conclusions derived from related studies with limited scope or purpose. Key functional areas were categorized according to their potential for shaping the in-space assembly strategy. Subsequently, available technical exploration study data related to these functional areas were perused for implementation options, issues, and trade study opportunities. The results were then synthesized into an option tree which will be expanded or contracted as required to coverage on a list of assembly implementation alternatives which may be comparatively evaluated to arrive at the preferred (or best alternative) assembly strategy for any case study.

Using the FY 1988 activity as a foundation, the FY 1989 priority will be directed towards defining the major assembly-related issues, performing work that leads towards an understanding of the assembly problem, and developing methodologies, criteria, and parameters that can be applied to evaluation, extrapolation, and/or derivation of alternatives. Although there exist many considerations that have an interaction with assembly, they will fall into two categories - those that are allowed to be driven by the assembly strategy and requirements, and those that tend to put requirements and constraints upon the assembly concept. A combination of management policy and sensitivity and assessment analyses will be required to place these considerations in the proper category. It is viewed as necessary to focus the various activities that affect assembly and to be able to steer or be steered by technical considerations. Therefore, an assembly development team consisting of participants from the various integration activities should be established. Since many related study results may be based on specific focused objectivities, it will be necessary to implement an end-to-end assembly analysis activity to assess whether implementation of recommendations across the integration agents is compatible with assembly.

It is recognized that there is a trade between onorbit assembly and ETO launch capability. Launch vehicle performance and flight rate are major assembly-shaping parameters in that they drive how the space transfer vehicles can be packaged and the time required to complete assembly; these factors in turn drive lower level technical requirements for onorbit support. Since one of the considerations that must be involved as part of the tracking between onorbit assembly approaches and ETO launch capability is the monetary cost associated with implementation of requirements, a proposed approach was conceived to develop a method for ascertaining OEXP requirements for launch performance. This proposal, its validity as yet unsubstantiated, is based upon total cost associated with ETO development, operational support cost, and number of ETO vehicles to be produced. An FY 1989 pursuit will be to determine if these considerations can be correlated to ETO delivery capability and thus if a cost-optimized ETO payload-to-orbit capability can be derived. Having established this parameter (or range), the effect on the in-space elements and operations would then need to be established, thus allowing an assessment (and perhaps adoption) of one assembly parameter.

It should be a management goal to steer the study and results in such a manner that conclusions can be drawn and generalized to the maximum extent. This goal is intended to provide a sufficiently wide and visionary perspective of the assembly process and the associated key parameters to allow informed choices of extrapolation of future assessments of changes in requirements without total restudy of the problem.

Findings. A preliminary list of assembly-strategy-shaping parameters is contained in table 4.4.1-I. Determining the degree to which assembly objectives, required capabilities, and imposed constraints should be allowed to drive (rather than be responsive to) mission objectives and vehicle configurations will require a methodical and integrated approach.

Potential assembly strategy-related options, issues, and trade studies are presented in figure 4.4.1-1, including identification of a candidate assembly case study for initial FY 1989 analysis activities. This case study may be revised as FY 1989 activity matures.

4.4.1.2 Assembly of Phobos Spacecraft in LEO

The objective of this study is to evaluate the assembly of the Phobos mission vehicle in LEO. Onorbit mating of multiple elements of a vehicle becomes more and more practical as the size of the pieces being lifted to LEO increases.

Background. A variety of previous studies of transportation node space stations has concentrated on the problems of assembling, refurbishing, and maintaining fully or partially reusable transportation systems for translunar or trans-Mars manned flight. This previous work has concentrated on long-term scenarios that assume a substantial human presence in LEO. Onorbit assembly and test of spacecraft, as well as cryogenic propellant transfer and storage, are in general assumed. Recently proposed piloted Phobos and Mars missions have assumed this capability.

On the other hand, the infrastructure and many of the technologies needed to assemble, test, and launch large spacecraft from LEO do not exist at present, a lack which poses an obstacle for proposed missions carrying humans to Mars with Earth departure dates on or around the year 2000. The Space Station Freedom Program's Evolution Working Group is now working to characterize the projected use of the phase II station. Among the

TABLE 4.4.1-I.- END-TO-END ASSEMBLY CONSIDERATIONS

1. Assembly cost
    a. Development (ETO, infrastructure, mission)
    b. Production
    c. Operations

2. ETO transportation capabilities
    a. Performance (PL mass, size)
    b. Flight rate
    c. Reliability

3. Space transfer vehicle
    a. Launch package elements (mass, functionality per launch)
    b. Assembly and checkout requirements
    c. Maintenance and vehicle support during assembly
    d. Reusability leverage

4. Mission requirements
    a. Mass delivered to LEO
    b. Departure date/LEO departure window
    c. Assembly frequency

5. Node support
    a. Functions/services provided
    b. Modifications
    c. Operations evolution and applications

6. Assembly tasks and functions
    a. Crew resources available
    b. Duration requirements and constraints
    c. Sequencing
    d. Assembly location/facility (e.g., SS, STS)
    e. Vehicle assembly and support systems assembly
    f. Checkout

7. Orbital resources and environment
    a. Vehicle requirements, interim configuration, (power, crew, thermal, data, etc.)
    b. Orbit and orientation (departure requirements, orbit-keeping strategy, sun angles, contamination, debris, launch vehicle performance, etc.)
    c. Available accommodations (including manned provisions)

8. Logistics
    a. Maintenance/repair during assembly
    b. Spares, tools, and expertise location and availability (space/ground)
    c. Transportation (ETO, orbit-to-orbit, etc.)
    d. Consumables loading
    e. Onorbit maintenance, refurbishment, and turnaround (disassembly/assembly for subsequent missions)

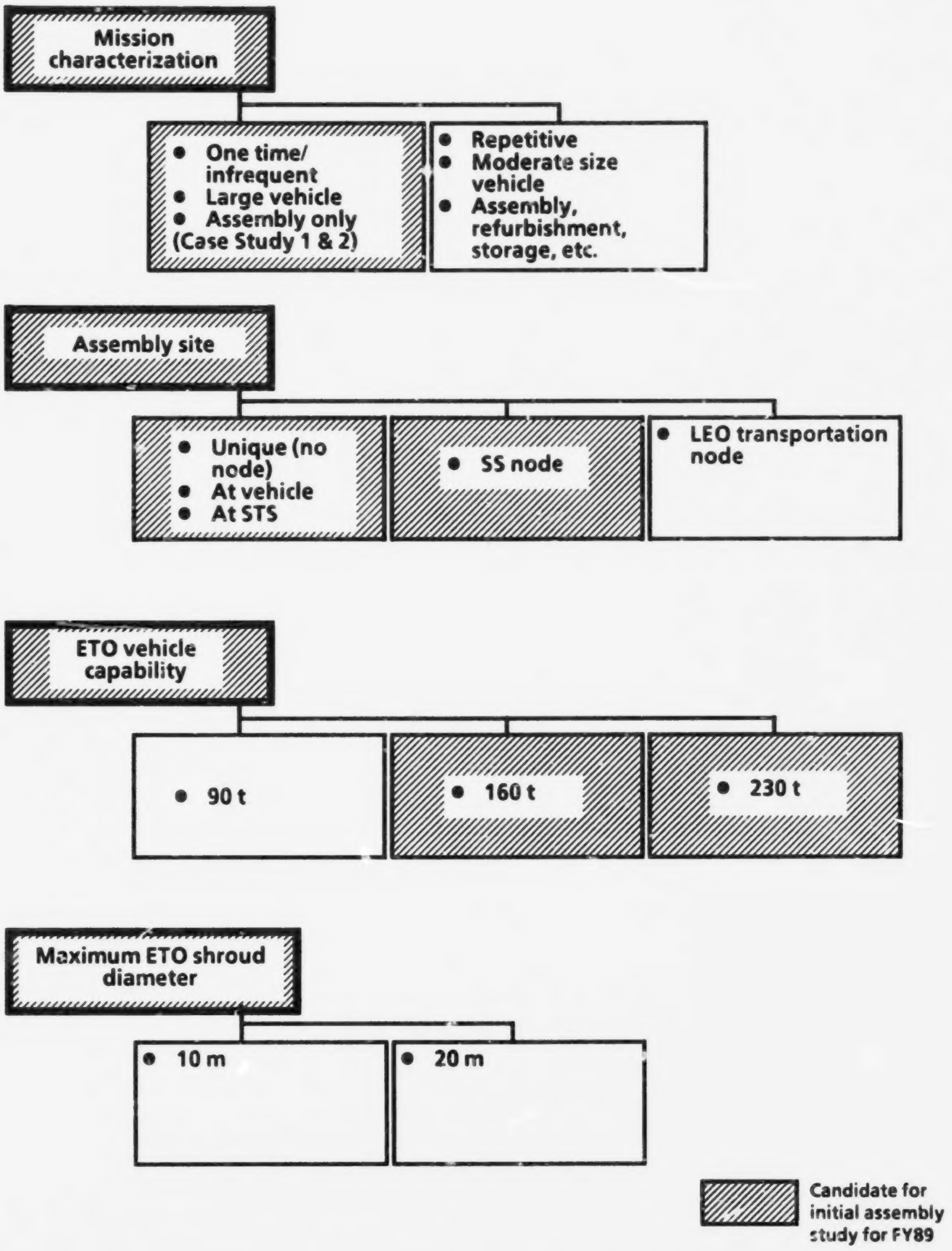

Mission characterization
• One time/ infrequent
• Large vehicle
• Assembly only
(Case Study 1 & 2)
• Repetitive
• Moderate size vehicle
• Assembly, refurbishment, storage, etc.
Assembly site
• Unique (no node)
• At vehicle
• At STS
• SS node
• LEO transportation node
ETO vehicle capability
• 90 t
• 160 t
• 230 t
Maximum ETO shroud diameter
• 10 m
• 20 m
Candidate for initial assembly study for FY89

Figure 4.4.1-1.- LEO assembly strategy, inputs, studies, and trades -- preliminary.

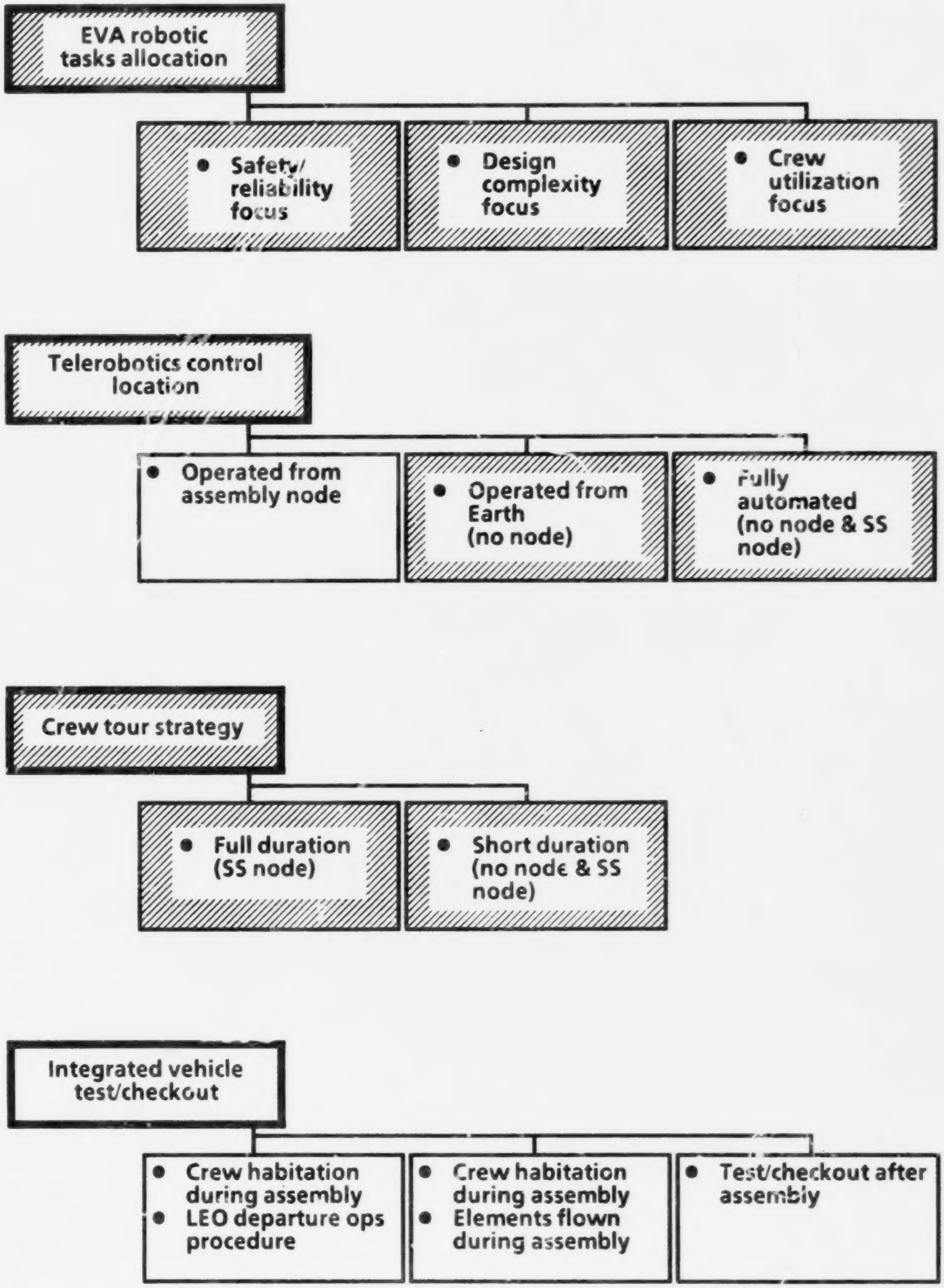

EVA robotic tasks allocation
• Safety/ reliability focus
• Design complexity focus
• Crew utilization focus
Telerobotics control location
• Operated from assembly node
• Operated from Earth (no node)
• Fully automated (no node & SS node)
Crew tour strategy
• Full duration (SS node)
• Short duration (no node & SS node)
Integrated vehicle test/checkout
• Crew habitation during assembly
• LEO departure ops procedure
• Crew habitation during assembly
• Elements flown during assembly
• Test/checkout after assembly

Figure 4.4.1-1.- (Continued).

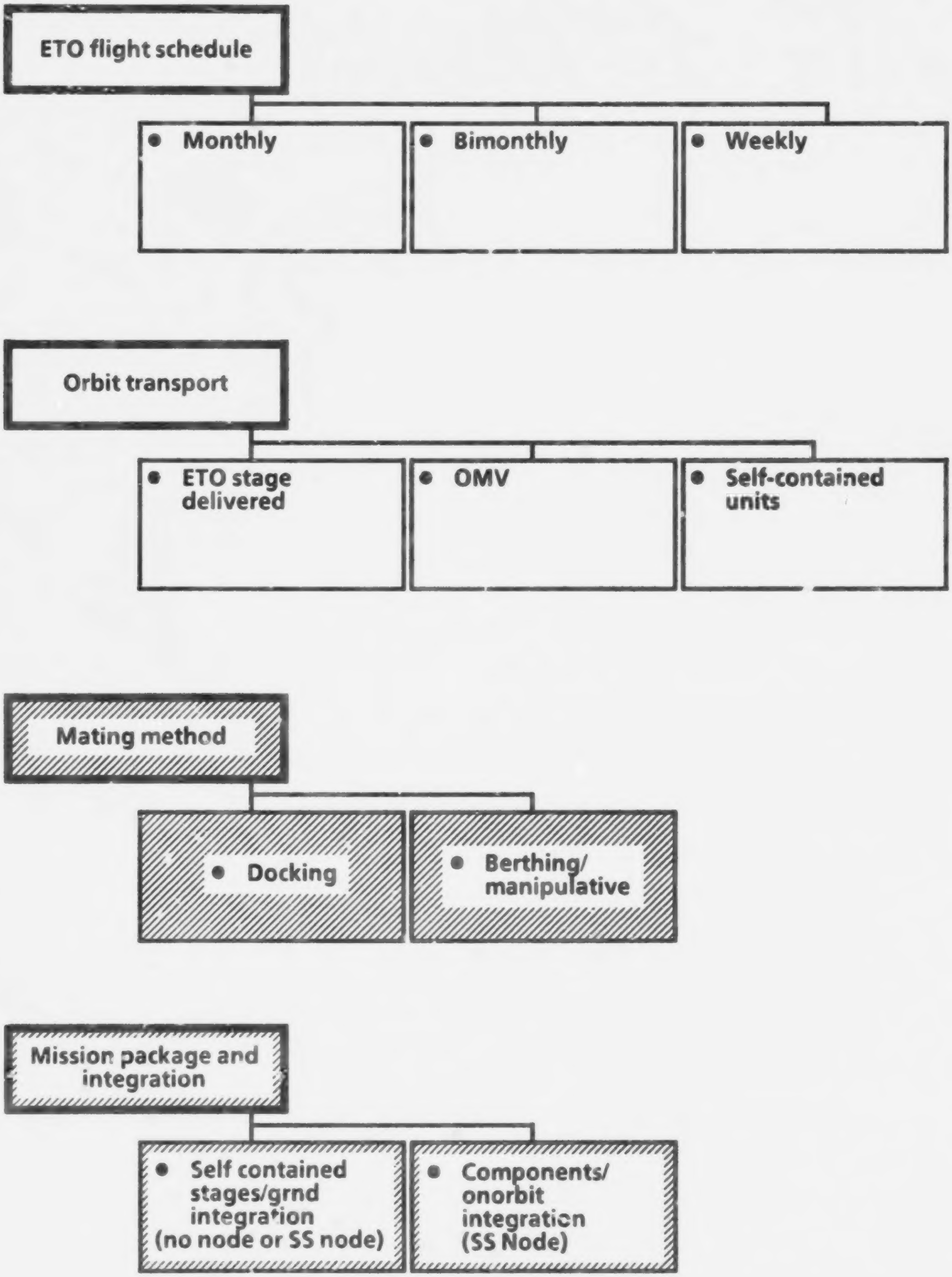

ETO flight schedule
Monthly
Bimonthly
Weekly
Orbit transport
ETO stage delivered
OMV
Self-contained units
Mating method
Docking
Berthing/ manipulative
Mission package and integration
Self contained stages/grnd integration (no node or SS node)
Components/ onorbit integration (SS Node)

Figure 4.4.1-1.- (Continued).

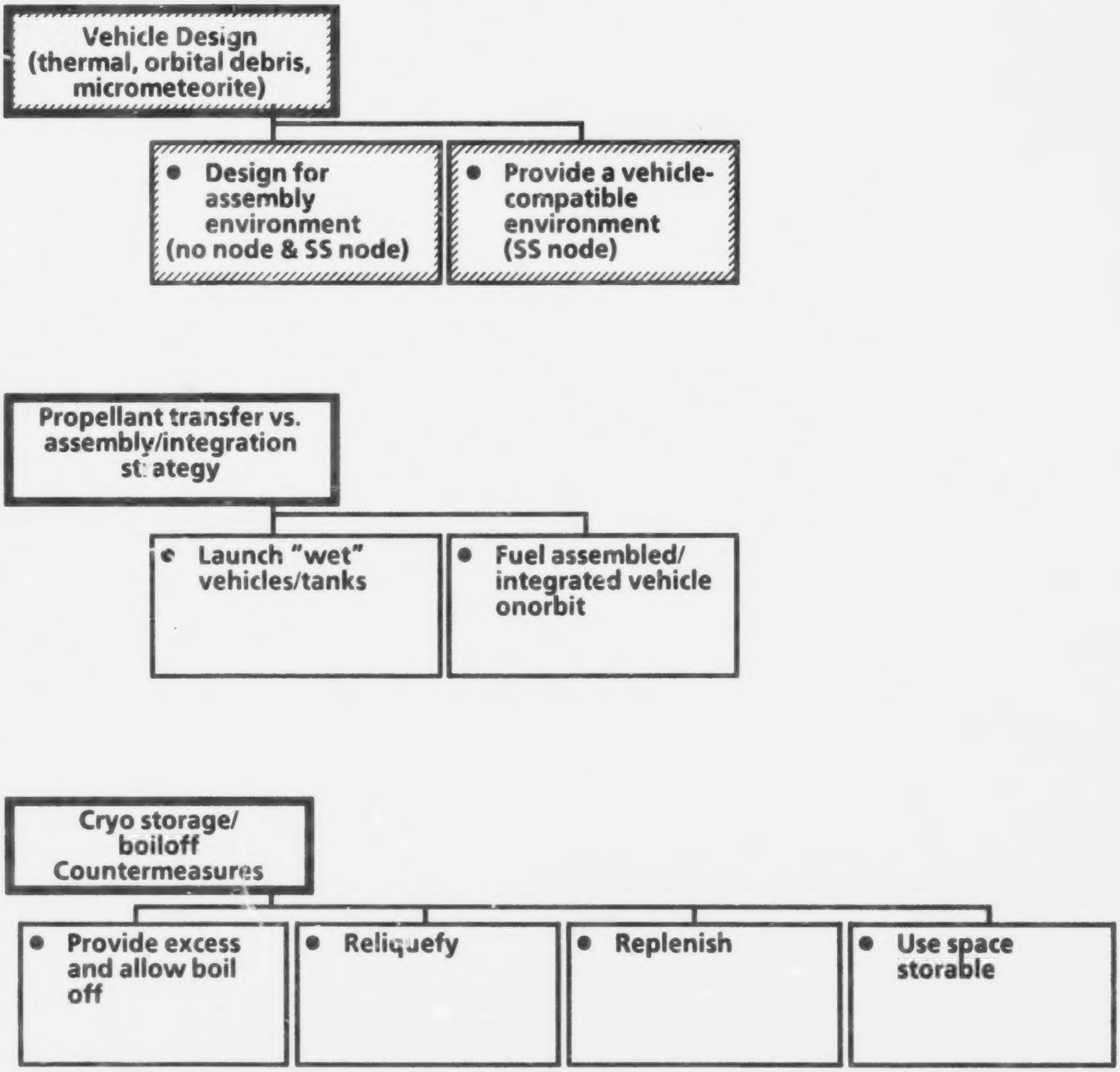

Vehicle Design (thermal, orbital debris, micrometeorite)
Design for assembly environment (no node & SS node)
Provide a vehicle-compatible environment (SS node)
Propellant transfer vs. assembly/integration strategy
Launch "wet" vehicles/tanks
Fuel assembled/integrated vehicle onorbit
Cryo storage/boiloff Countermeasures
Provide excess and allow boil off
Reliquefy
Replenish
Use space storable

Figure 4.4.1-1.- (Concluded).

important issues being addressed is how Space Station Freedom might evolve into a transportation node in support of exploration-class missions.

It is crucial to long-term program planning to closely examine the technical, operational, and scientific research ramifications of using the station to support assembly of large space transfer vehicles in the 1000-t range. From this study a determination will be made as to whether it would be advantageous to branch to a second LEO node and when to do so.

Additionally, OEXP studies are investigating the potential for assembling large space vehicles in LEO without the use of a transportation node or other space-based infrastructure of significance. The Phobos vehicle assembly concepts discussed in this section initially assumed no additional LEO infrastructure.

Key Assumptions.

1. The Phobos configuration (from Martin Marietta viewgraph package CS-1.MMSS-1, "Human Expedition to Phobos, Case Study 1: Transportation," 7/10/88) was used for assessment.
2. The ETO capability assumed for delivery of the Phobos vehicles (cargo and piloted) was an advanced launch system (ALS)-class (96-t) launcher.

Approach. The approach employed for this activity was to evaluate the onorbit assembly concept of the Martin Marietta Phobos vehicle configuration, to identify issues and/or areas of concern, and to recommend alternatives for incorporation into the FY 1989 Expedition to Phobos Case Study requirements.

Findings. Figure 4.4.1-2 shows the Martin Marietta Phobos vehicles (cargo and piloted) to be assembled. Numerous large fluid connections are required. Eleven large tanks come together to make up the trans-Mars injection (TMI) single stage. A summary of selected key assembly requirements for this vehicle is shown in table 4.4.1-II. Extravehicular activity (EVA) was estimated by determining each task to be performed for each tank or stage brought up. The estimate arrived at was then

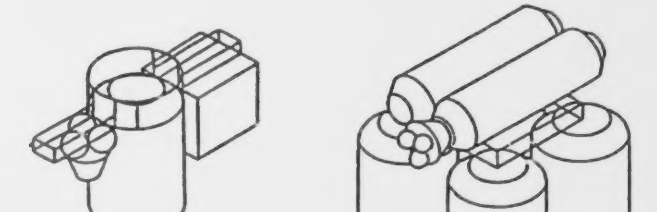


Figure 4.4.1-2.- Assembled vehicles, cargo and piloted.

TABLE 4.4.1-II .- SUMMARY OF PHOBOS VEHICLE ASSEMBLY REQUIREMENTS, MARTIN MARIETTA VEHICLE

Item	(Martin) Vehicle
Total LEO mass at dep., both veh. (t)	2,096
Cargo veh. LEO mass (t)	467
Piloted veh. LEO Mass (t)	*1,310
No. of HLLV launches req. for both veh.	24
No. of STS launches req. for both veh.	5
HLLV max. payload to LEO required (t)	96
HLLV max. shroud dia. req., meters (ft)	10 (33)
HLLV max. shroud length req., meters (ft)	30 (100)
EVA req. for cargo veh. assem., no. of 6-hr EVA's	56
EVA req. for piloted veh. assem., no. of 6-hr EVA's	131
Total EVA req., no. of 6-hr. EVA's	187
No. of req. large dia. (20-inch) fluid line connections	32
No. of req. small dia. fluid line connections	96
No. of req. structural in-space connections	117
No. of req. electrical in-space connections	28

*Becomes 1180 if MOO2 burn prop. is moved to cargo veh.

multiplied by a 1.5 factor as a margin for things forgotten or unknown. The true value is highly influenced by the level of technology used in assembly and is difficult to accurately estimate at this level of detail. The objective at this time is to develop a reference and method to enable comparisons at an order-of-magnitude level of accuracy rather than to estimate exact values. Better definition of the hardware interfaces will be required to improve the EVA estimates, however.

Based on initial assessment of the explicit or inferred requirements, assembly of the reference Phobos vehicle without the use of some type of facility in space is, at best, questionable and cause for concern; at worst, not possible as proposed.

The Shuttle remote manipulator system (RMS) is rated for only a third of the 90-t tank mass that must be moved around. In addition, the Shuttle must dock at a variety of locations on the vehicle in order to use the manipulator to place the tanks, which, though possible, seems impractical. Twenty-five months are required to assemble the cargo and piloted vehicles, using an estimated 187 EVA's of 6-hour duration each, or approximately two EVA's for each week. The Orbiter fleet cannot support this length of stay or number of EVA's. However, it may be possible to use the habitation module of the piloted vehicle to support the EVA's, with some weight penalty on the whole mission. The Orbiter and RMS must be on hand to place each of the 24 payloads, however. Additionally, it is not viewed as practical to fly a 90-t tank into a slot between other tanks and position it with sufficient accuracy to make up eight fluid and four structural connections and at least one electrical connection. The tank must be positioned with a manipulator or other device rigidly connected to the vehicle. Thus it appears that a space facility will be necessary if assembly is to be constrained to use of current support system capabilities and concepts.

The preceding concerns led to the concept for assembly shown in figure 4.4.1-3 with an RMS capable of reaching any point on the vehicle requiring placement of a tank. The figure concept shows a manipulator and truss structure only. Power, thermal control, attitude control, habitation, and EVA/airlock/spacesuit support are all assumed to come from another source, most likely a space station rigidly attached to the structure.

The assembled vehicles (figure 4.4.1-2) are essentially put together piece by piece. An ALS-class launcher capable of placing the 90-t tanks in LEO is assumed. The RMS travels on a strongback. The vehicle is assumed to be docked to some rotating fixture that will allow the single RMS on a strongback access to it all. EVA or a capable robotic equivalent is required for numerous fluid, electrical, and structural interconnects. The biggest challenge is the 128 fluid connections that must be made in space, including 34 large-line (20-inch or so) connections. Bolted connections and leak tests were assumed to be required for these large lines. Quick connects similar to the Shuttle/external tank interface may also be possible.

Figure 4.4.1-4 shows the launch schedules for assembly of the Phobos vehicle and associated Shuttle support launches. Shuttle launches are required to replace assembly crews every 6 months or so. A minimum launch rate of one ALS-class stage per month is required to assemble the piloted vehicle between the cargo vehicle and the piloted vehicle departure dates, roughly 18 months. Shuttle launches concurrent with the ALS launches are also required.

During the course of this study and evaluation, questions arose of whether assembly can be simplified and what options may be available for consideration. Conceptually, it was judged that docking stages together without fluid connections might be a viable candidate configuration option to be levied as a vehicle study requirement. To confirm the potential merit, the Phobos vehicle TMI stage was scaled parametrically to estimate the effect of multiple stages on the mass of such a configuration, as well as the effect on the other assembly parameters (tables 4.4.1-III and -IV). It is recommended that ETO capabilities up to 230 t and a Phobos vehicle configuration that includes both the addition of an aerobrake and multiple TMI stage options be study requirements for FY 1989.

Issues. The following issues need to be considered for assembly:

1. How will the ALS payload and upper stage be brought to the assembly point? Should they fly themselves up to a docking structure or should an orbital maneuvering device dock with them and bring them to the assembly point?

2. What is the optimum-altitude assembly orbit? Are there requirements on the departure inclination? Is the phase 1 Space Station Freedom orbit adequate?

3. What is the micrometeoroid/orbital debris shielding requirement for these vehicles? Are multiwall shields required on the tankage?

4. What are the penalties associated with boiloff? Can it simply be vented, should it be captured and used for attitude control and orbit makeup, or should it be reliquefied and placed back in the tank? Is the capability to top off the tanks required?

5. How can the vehicles accommodate other launch opportunities with different delta V requirements?

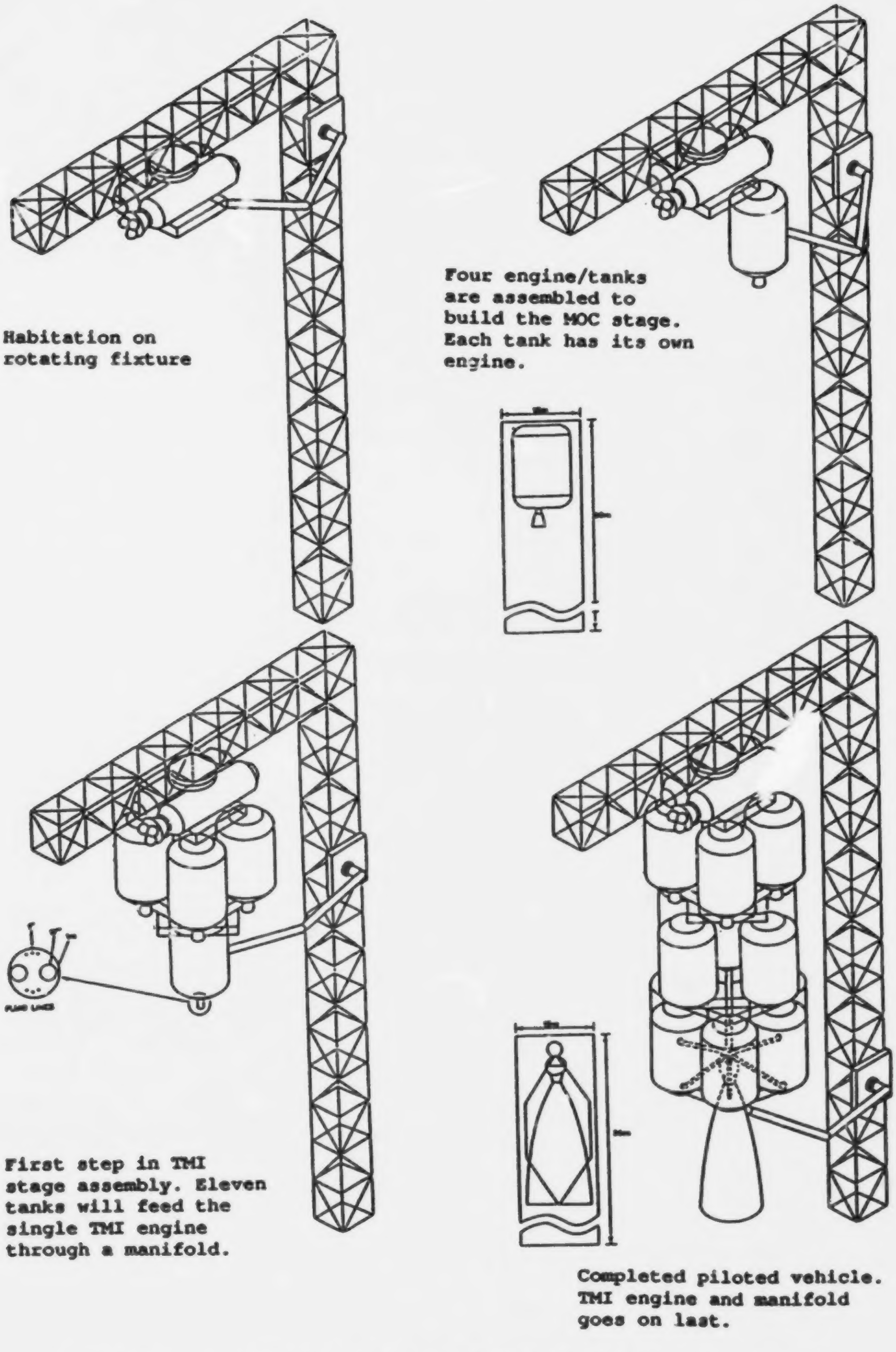
6. Current launch vehicles (manned and unmanned) have an ascent success rate of roughly 91 percent over 447 flights (includes all major U.S. and foreign launchers). This is simple ascent reliability. On-time performance is much worse, not even measured. Given this 1-in-10 failure rate and poor schedule performance history, should a vehicle requiring multiple launches prefer a few large launches or many small ones? How can we make the system insensitive to a launch vehicle failure?

7. What is the maximum time a crew can work onorbit, supporting two EVA's per week or more?

8. Can the assembled vehicle with over 100 fluid connections be launched from LEO without a hot-fire engine test for the TMI stage?

9. Can the assembled vehicle be adequately vibration-tested on the ground? Is a vibration test in space required?

10. Is it possible to build a large-diameter (20-inch) quick connect for cryogenic fluids that requires no leak testing with cryogens in it?



Habitation on rotating fixture

Four engine/tanks are assembled to build the MOC stage. Each tank has its own engine.

First step in TMI stage assembly. Eleven tanks will feed the single TMI engine through a manifold.

Completed piloted vehicle. TMI engine and manifold goes on last.

Fig 4.4.1-3.- Assembled (Martin Marietta) piloted vehicle assembly sequence.

Launch Number	Launch Vehicle	Payload	2000										2001												2002							
			Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug
1	ALS 1	TEIS, PHEV, Cargo				●																										
2	STS-1	Assembly crew #1 (AC #1)				●																										
3	ALS-2	MOS, MMOOS					●																									
4	ALS-3	TMIS (tank #1)						●																								
5	ALS-4	TMIS (tank #2)							●																							
6	ALS-5	TMIS (tank #3)								●																						
7	ALS-6	TMIS (tank #4)									●																					
8	ALS-7	SSME+										●																				
9	STS-2	Inspec. Crew #1 A.C #2												● ▨																		
10	ALS-3	HAB modules ECCV												●																		
11	ALS-9	MOCS (tank #1) MOO-1													●																	
12	ALS-10	MOCS (tank #2)] MOO-1														●																
13	ALS-11	MOCS (tank #3) MOO-1															●															
14	ALS-12	MOCS (tank #4) MOO-1																●														

▨ Launch Cargo Mission from LEO

Launch Number	Launch Vehicle	Payload	2000										2001												2002							
			Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug	Sep	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	Aug
15	ALS-13	TMIS (tank #1)																	●													
16	STS-3	A.C. #3; A.C. #2 returns																	●													
17	ALS-14	TMIS (tank #2)																		●												
18	ALS-15	TMIS (tank #3)																			●											
19	ALS-16	TMIS (tank #4)																				●										
20	ALS-17	TMIS (tank #5)																					●									
21	ALS-18	TMIS (tank #6)																						●								
22	ALS-19	TMIS (tank #7)																							●							
23	STS-4	A.C. #4; A.C. #3 returns																							●							
24	ALS-20	TMIS (tank #8)																								●						
25	ALS-21	TMIS (tank #9)																									●					
26	ALS-22	TMIS (tank #10)																										●				
27	ALS-23	TMIS (tank #11)																											●			
28	ALS-24	SSME+																												●		
29	STS-5	Flight crew ins. cr. (on shut)																													●	▨

▨ Launch Human Mission from LEO

Figure 4.4.1-4.- Schedule for assembled (Martin Marietta) vehicle.

TABLE 4.4.1-III.- SIZING A STORED TMI PHOBOS VEHICLE

Piloted Vehicle:					
No. of TMI stages **	1	2	3	4	5
Total LEO mass (t)	1,180	959	925	867	865
TMI stage wet mass* (t)	877	352	223	156	124
Cargo Vehicle:					
No. of TMI stages	1	2	2	2	3
Total LEO mass (t)	664	621	621	621	611
TMI stage wet mass* (t)	454	205	205	204	132

* All TMI stages are the same size, to reduce manufacturing costs and make maximum use of the launch vehicle. All TMI stages use a 10-m maximum diameter.

** The single-stage TMI vehicles assume the MOO2 burn propellant has been transferred to the cargo vehicle, as suggested by Martin Marietta, making the piloted vehicle 130 metric tons lighter than the Martin Marietta Phobos vehicle.

TABLE 4.4.1-IV.- SUMMARY OF PHOBOS VEHICLE ASSEMBLY REQUIREMENTS, DOCKED VEHICLE

Item	Docked Vehicle
Total LEO mass at dep., both veh. (t)	1,546
Cargo veh. LEO mass (t)	621
Piloted veh. LEO mass (t) M002 propel. on cargo veh.	925
No. of HLLV launches req. for both vehicles	8
No. of STS launches req. for both veh.	4
HLLV max. payload to LEO required (t)	226
HLLV max shroud dia.req., meters (ft)	10 (33)
HLLV max shroud length req., meters (ft)	44 (145)
EVA req. for cargo veh. assem. (6-hour EVA's)	7
EVA req. for piloted veh. assem.(6-hr EVA's)	10
Total EVA req., (6-hr. EVA's)	17
No. of req. large-dia. (20-inch) fluid line connections	0
No. of req. small-dia. fluid line connections	0
No. of req. structural in-space connections	36
No. of req. electrical in-space connections	7

11. How can low-level leak testing be done in space? Gas sniffers will not work in a vacuum. Is it practical to run a spectrometer device over the surface of all the plumbing or each connection? Are leaks in the TMI stage plumbing really important in space?

12. What is the level of complexity of the plumbing for the TMI stage made up of 22 tanks? What is the minimum number of fluid connects required per cryogenic tank?

13. Is there a docking concept that would allow the assembled vehicle to be put together without a manipulator to move the tanks around?

14. What mass of facilities and consumables is required to support two EVA's per week over a 2-year period?

15. When is it reasonable to establish assembly parameter allocations on the transportation and node allocations, and what is the path to reasonable allocations?

16. Can an assembly facility coorbit with the phase 1 Space Station Freedom without unreasonable propellant-use penalties?

4.4.2 Onorbit Assembly vs. Ground Assembly Functions

The objectives of this trade study are to determine the spacecraft assembly, fueling, and verification tasks required onorbit and to establish the method (EVA, automation, and/or teleoperation of robotic and semi-automated equipment) for performing these tasks.

Ground rules. The following ground rules will be used in the studies of onorbit assembly/verification requirements:

a. LEO node operations for assembly/verification/maintenance of New Initiative vehicles will have processes similar to current ground activities for such tasks.

b. All operations that can be done on the ground will be.

c. Flight elements will be fully tested before launch to LEO.

d. Flight software will be verified before insertion into onboard computers.

e. EVA will be used only when necessary.

f. Mission vehicles will be designed and built to facilitate onorbit assembly.

g. The LEO transportation node is assumed to be the phase 1 Space Station Freedom.

Approach. In order to determine the requirements for onorbit assembly and verification of vehicles, the job must be described in terms of its component tasks. Most historical data on the final assembly of spacecraft are found in the flows constructed for assembly of current and past vehicles at the launch center. On the assumption that many of the tasks in these flows are applicable whether the job is done on the ground or onorbit, the starting point for the trades is to collect flows for ground processing of spacecraft such as the STS, Delta, and Apollo (in particular the manned module).

To determine which tasks in the current processing flows must be done onorbit: each task must be assessed for its applicability to future vehicles, and a set of criteria must be defined to establish the necessity for onorbit processing of each task or set of tasks in the flows. For example, a critical function will be to verify electrical and mechanical connections after any assembly process onorbit, even though this may be accomplished in a manner different from current ground checks. The criteria to determine if a task or a set of tasks must be done in LEO follow:

a. If the physical limitations of the ETO carriers preclude assembly of the vehicle on the ground, onorbit assembly will require certain tasks.

b. Given the above condition, tasks associated with transport and receiving of equipment will be required onorbit.

c. Tasks that must be done subsequent to assembly of any two or more elements will be onorbit tasks.

The following sets of tasks, which will be expanded and defined in more detail in the final study results, are required onorbit. This is only a partial listing.

a. Any "pieces" brought to orbit for assembly require receive-and-inspect operations.

b. If an expendable launch vehicle or heavy-lift launch vehicle (ELV/HLLV) brings parts to be assembled, the parts must be moved to the node (probably by orbital maneuvering vehicles (OMV's)); thus transport of vehicle elements is a required set of tasks.

c. Onorbit assembly requires onorbit verification of electrical and mechanical interfaces.

d. Some flight software must be loaded after assembly.

e. Final installation of any hazardous materials (e.g., ordnance, nuclear energy sources) must be done off-node and as close to stage ignition as possible.

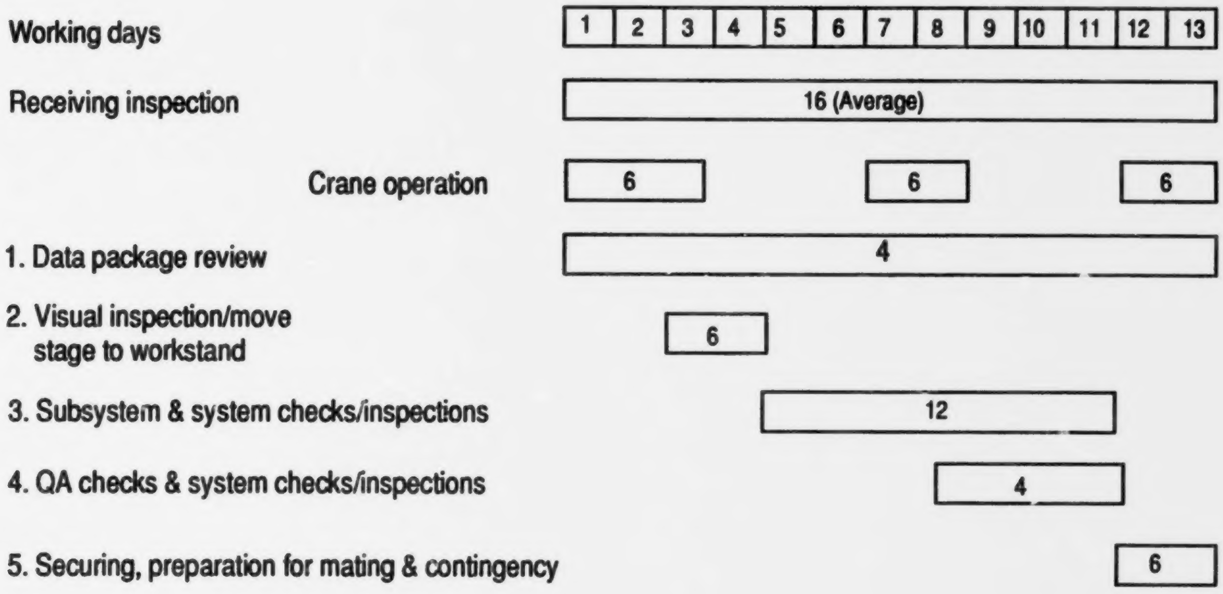
f. Final verification test of all systems, final diagnostic checks, and state vector update must be done as close to stage ignition as possible.

g. Where practical, propellant loading is done as close to stage ignition as possible.

h. Vehicle must be moved to an off-node position and stabilized prior to stage ignition.

In conjunction with the definition of onorbit tasks, the equipment to support those tasks must be defined. This onorbit support equipment (OSE) will be much the same as the equipment used in the current ground processes; however, some of the ground support equipment will probably have to be revised for in-space use and some may be replaced by automation.

Findings. Figures 4.4.2-1 through 4.4.2-3 are representative of the flows being refined to describe



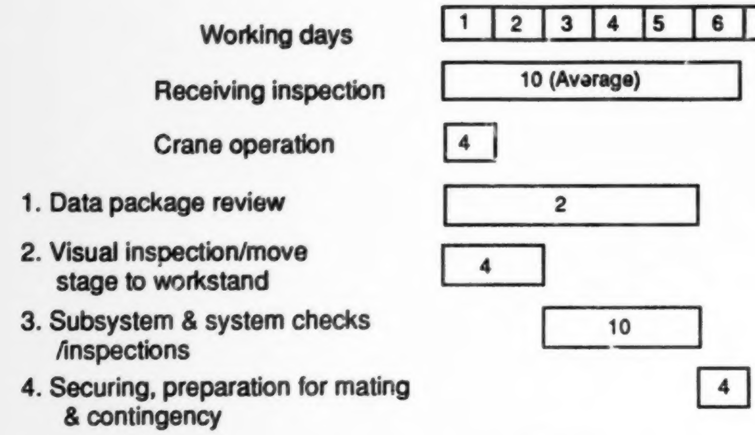
Total # man hours=(Approx) 1500 man hours/stage (One eight hour shift per day)

Skill mix	Facility	Resources
Engineers 4 Technicians 6 Inspectors 4 Others 2 Total 16	Floor space 38,000 sq. ft. (Present delta floor space)	Power-gases-fluids-cranes spares-lighting-air conditioning

GSE Requirements=155 separate pieces of equip. & computer room/small control room set up

GSE Bottom Line
Long list of GSE available upon request

Figure 4.4.2-1.- Receiving inspection—1st stage. Past/present ground processing—2 stage cryogenic vehicle + crew module & aerobrake



AI, robotics, and built-in test equipment could improve ground processing time up to 50% or more (Delta at present expends about 1500 man hours for this effort and twice as long.)

Total # man hours=500-600 (One eight hour shift/day)

Skill mix		Facility	Resources
Engineers	3	Floor space 38,000 sq. ft. (Present delta floor space)	Power-gases-fluids-cranes spares-lighting-air conditioning
Technicians	5		
Inspectors	1		
Others	1		
Total	10		

GSE Requirements =

- Slings
- Meters
- Pallets
- Handling devices
- Automatic test equip
- Pressure test equip
- Electrical equip
- Hydraulic equip
- Hand tools

GSE Bottom line
Delta currently requires 155 pieces of GSE for this operation ranging from meters to main frame computers.

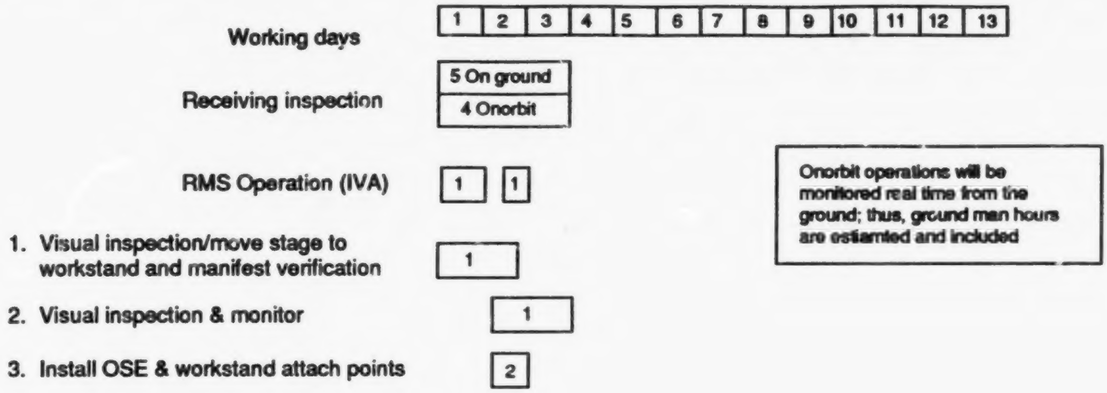
Figure 4.4.2-2.- Receiving inspection—1st stage. Improved ground processing—2-stage cryogenic vehicle + crew module & aerobrake.

onorbit operations. These figures cover only the tasks required for receiving/inspection of spacecraft elements.

The processing of spacecraft on the ground in the current flows requires several weeks and several hundred people for each vehicle. Since this is not possible for onorbit operations, it is obvious that things must be done differently during LEO transportation node operations.

It must be understood that the level of man loading onorbit for receiving/inspection in figure 4.4.2-3 is dependent on the use of different vehicle designs to facilitate in-space assembly and on the use of automation and telerobotics. To date, these trades simply assume that many tasks can be automated and/or done by the use of teleoperation and artificial intelligence (AI). Such assumptions will require considerable study to assess their validity and to determine the design characteristics necessary to facilitate the use of automation and teleoperation.

The determination of how the described tasks are to be done will show that, in some cases, automation is enabling technology; i.e., if the tasks cannot be automated they must be deleted from the flows and a workaround devised. In other cases, the use of automation and teleoperation allows a simplification of the tasks and can substitute for the use of intravehicular activity and EVA crewtime; therefore, the technologies are considered to be enhancing. For example, AI can be used to offload crewtime in the decision processes associated with equipment troubleshooting. In all cases, the trade of crewtime for automation and robotics must be assessed from the standpoint of ability to design the equipment necessary to automate the tasks.


Working days
1
2
3
4
5
6
7
8
9
10
11
12
13
Receiving inspection
5 On ground
4 Onorbit
RMS Operation (IVA)
1
1
1. Visual inspection/move stage to workstand and manifest verification
1
2. Visual inspection & monitor
1
3. Install OSE & workstand attach points
2
Onorbit operations will be monitored real time from the ground; thus, ground man hours are estiamted and included

Total # man hours = 36 IVA hours
8 hours (AI & robotics may reduce EVA further by as much as 50%)
+120 ground hours (estimated that a minimum of 5 experts on ground will monitor all OPS)

Skill Mix		Onorbit Facility	Onorbit Reserves
Onorbit	Ground		
IVA Mission specialist 2 EVA Mission specialist 2 Total 4	Engineers 3 Technicians 1 Others 1 Total 5	Control area plus floor space 10,000 sq. ft. (unpressurized)	Power-gases-fluids RMS-spares-lighting

OSE Requirements=OSE/RMS interface connectors (quick connect interfaces)
Automatic test equipment
Teleoperated test equipment
Power assisted hand tools
Data pack loaded & integrated with OSE and on-board computer

GSE Bottom line

Automation/robotics/teleoperated sensors & monitoring devices must be used as much as possible. 24 hour monitoring of all test and operations from both ground and internal to space station is essential.

Figure 4.4.2-3.- Receiving inspection–1st stage. Mandatory onorbit minimum activity processsing–2 stage cryogenic vehicle + crew module & aerobrake.

Figure 4.4.2-3.- Receiving inspection—1st stage. Mandatory onorbit minimum activity processing—2-stage cryogenic vehicle + crew module & aerobrake.

4.4.3 LEO Assembly Operations Support Systems and Techniques

The objective of this study is to assess the operational methods and techniques for LEO assembly.

Assembly operations for Human Expeditions to Mars (Case Study 2) are representative of assembly operations required by exploration missions using similar technology enroute to other solar system destinations. Extrapolation of these results to assembly operations at non-LEO sites in space is also expected to be possible.

Key Assumptions. Key assumptions and requirements extracted from the SRD are as follows:

a. Case Study 2 will be assessed.

b. A LEO node is used to support assembly operations.

c. Multiple manned missions to Mars are to be supported.

d. Each mission consists of an unmanned cargo flight and a piloted flight.

e. All space transfer vehicles are expendable and employ chemical propulsion systems.

f. Advanced technology requirements for complex assembly support systems are to be minimized.

Approach. The LEO assembly phase of the Human Expeditions to Mars Case Study will be used for an initial operations assessment. A set of operational guidelines for LEO assembly will be developed to support evaluation of the current proposed designs for the appropriate systems and the assumptions from which they were generated. Aspects of the design philosophy which negatively impact assembly operations will be identified.

An interactive relationship with all OEXP participants involved in this activity will be established and maintained to support development of an operationally realistic vehicle/node design and candidate assembly scenarios, the objective being to expose more subtle, operationally-driven system design requirements. Preliminary data requirements needed to assess the assembly scenarios have been identified and are presented in table 4.4.3-I. These OEXP scenarios will be used to support the

TABLE 4.4.3-I.- LEO ASSEMBLY OPERATIONS ASSESSMENT INPUTS

Assembly Element	Preliminary Data Requirements
LEO node	Assembly sequence(s) - Scenario overview: event description/timeline
	Assembly support equipment (e.g., freeflyers, manipulators, EMU, MMU)
	Node configuration definition(s) - Physical layout - Systems capabilities - Orbit
	Propellant storage facilities
	Crew complement - Space station crew - LEO support (assembly) crew
	Scheduling guidelines
ETO transportation	Manifest - Cargo element definition/sequence - Flight schedule - Flight support equipment
	Performance - Lift capability - Margin/reserves allocation - Orbital operations capabilities (e.g., survival lifetime, controllability)
	Profile description - Event timeline
Space transfer vehicle	Vehicle configuration throughout assembly
	Service support requirements - Utilities (e.g., power, thermal, system monitoring) - Maintenance - Assembly - Propellant fueling

identification of candidate tasks for automation and robotics (A&R); establish requirements on the assembly node for the placement and capabilities of manipulator systems; and identify EVA, freeflyer, and other support equipment requirements necessary to accomplish the assembly task.

Products. Two major products resulted from FY 1988 activities. A preliminary methodology by which additional case study elements may be evaluated for operations support systems and techniques has been developed. The process is iterative, initially supporting operations feasibility assessments of proposed assembly scenarios that eventually lead to the development of preferred operational methods and techniques for LEO assembly.

This methodology is being applied to LEO assembly operations for Case Study 2 and is using ancillary studies such as the "Manned Mars Mission Accommodation--Sprint Mission" by Langley Research Center (LaRC). Although only portions of the case study have been examined to date, interim results obtained in FY 1988 include the following preliminary products: operations task breakdown, assessment criteria/categories and goals, evaluation matrices, implementation alternatives, schedule of activities, assessment tools, and FY 1989 study candidates.

Findings. The LEO node will strongly influence the character of assembly operations. The amount of functional support this facility provides to both the vehicle being assembled and the assembly process itself will be a major factor in the overall design of the process.

The requirements for the ETO transportation needed to support the human expeditions to Mars are very demanding, particularly for operations support. Current studies indicate that a large number of flights, at a relatively high flight rate, will be needed to launch the space transfer vehicle assembly elements and the required propellants. The high flight rate is anticipated to impact virtually all major operations phases including launch vehicle processing and cargo integration, flight planning and reconfiguration, launch preparation, and launch and mission support. The use of multiple launch systems, such as an unmanned HLLV and the Space Shuttle, can be expected to introduce additional complexities.

The assembly of the space transfer vehicle will require the development of new operations support systems and techniques. Although advanced technology development requirements are to be minimized, they may be imposed if they hold promise of significant gains in productivity or reduction in mission risk. To date, no onorbit activity has demonstrated the kinds of operations this task will involve. Depending on the design of the vehicle, however, many of the required LEO assembly operations techniques may be developed and demonstrated by the Space Station Freedom program.

An area in which the Space Station Freedom program has already demonstrated an operational constraint is logistics resupply. The program analyses indicate that assembly requirements will be a function, rather than a driver, of the assembly logistics flight rate and ETO capability. ETO transportation program considerations, such as schedule, manifest, and ground logistics capabilities, will control the number and frequency of logistics flights to the LEO node. All onorbit operations will ultimately have to conform to this scheduling constraint.

The functional requirements on the LEO node may be driven by another schedule. Program schedules for the Human Expeditions to Mars Case Study show periods of time, between the launch of one vehicle and the beginning of the assembly process on the next, in which no assembly will be taking place. During these periods, operational considerations may preclude a manned presence throughout the functional lifetime of the node. The possibility exists, therefore, that the LEO node might have to function in both a permanently manned and a man-tended mode. This dual capability requirement could have a strong impact on the node systems design.

Issues/Open Items. Several open items exist for LEO assembly operations. Before any meaningful, indepth assessment can be performed, the operations capabilities and limitations of the systems involved must be defined. In addition, while the high level assembly scenarios developed so far indicate the use of an OMV, it is doubtful that this vehicle, in its current design configuration, will be capable of handling the tasks to which it is being assigned. Therefore, a set of requirements for a more robust freeflyer system may have to be defined.

Another open item which will have a major impact on operations is the location of the LEO support crew base. System functionality requirements will differ significantly depending on whether the assembly crew is based at the Space Station Freedom and has to be ferried to the assembly node, at the LEO assembly node facility itself, or in the manned Mars vehicle. If the crew is based at the station or in the Mars vehicle, the LEO node will have to be designed as a man-tended system. Basing the crew at the assembly node itself will require permanently manned capability.

Analysis of the Human Expeditions to Mars Case Study indicates that a significant number of the required ETO transportation flights will be dedicated to propellant delivery. At this time, the propellant handling/transfer systems and storage location have not been defined. Since the fueling process will have a major impact on the assembly and flight preparation operations, this lack of definition is an important open item.

Planned or Required FY 1989 Activity.

a. Continue LEO assembly operations assessment for Case Study 2.

b. Conduct orbital assembly operations assessment for other case studies.

4.5 POWER AND PROPULSION PARAMETERS FOR NUCLEAR ELECTRIC VEHICLES

The objective of this study is to provide a common and consistent set of projected nuclear propulsion vehicle performance parameters for use in the Office of Exploration (OEXP) case studies. Currently there is no known source to which one can refer when modeling a nuclear electric vehicle, especially for the Mars round-trip mission.

Background. Lewis Research Center (LeRC) received a request from Marshall Space Flight Center (MSFC) to supply common and consistent projected nuclear electric propulsion performance data. These data would be used to model the vehicles in the Case Study 4 missions. The requested data included variation of thruster efficiency with Isp for ion and magnetoplasmadynamics (MPD) systems and specific mass of the nuclear reactor system. The Space Propulsion Technology and Power Technology Divisions at LeRC formulated reasonable projections of current technology for the 2005 to 2010 era.

Approach. LeRC compiled component performance projections from the two principal technology divisions named above. Given the data in tabular form, some further analysis was performed using approximate software to either model or curve fit the data to existing technology models. In the case of the MPD thruster efficiency model, Mr. Jim Gilland of Sverdrup Corporation derived a new expression specifically for this study.

Product. LeRC has established a data base of thruster performance and reactor specific mass. This data base has the concurrence of the Office of Aeronautics and Space Technology (OAST) Propulsion, Power, and Energy Division (Code RP) and will be published quarterly. It contains tables, graphs, recommendations, and explanatory text and appendices.

Findings. Typical curves of ion and MPD thruster performance are given in figures 4.5-1 through 4.5-4. Figures 4.5-5 and Table 4.5-I present data for SP-100 type reactor systems.

Planned or Required FY 1989 Activity. LeRC will publish a quarterly assessment of nuclear electric performance parameters to be distributed to the participants in OEXP studies after receiving the imprimatur of Code RP. LeRC will also include an assessment of vehicle performance for those case study missions that use or can use nuclear electric propulsion.

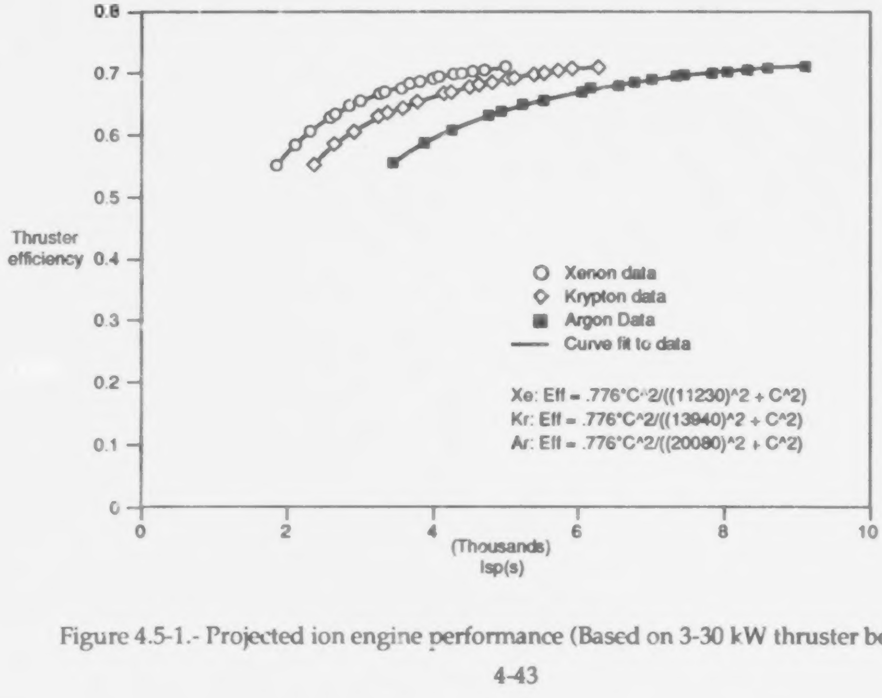


Figure 4.5-1.- Projected ion engine performance (Based on 3-30 kW thruster behavior).

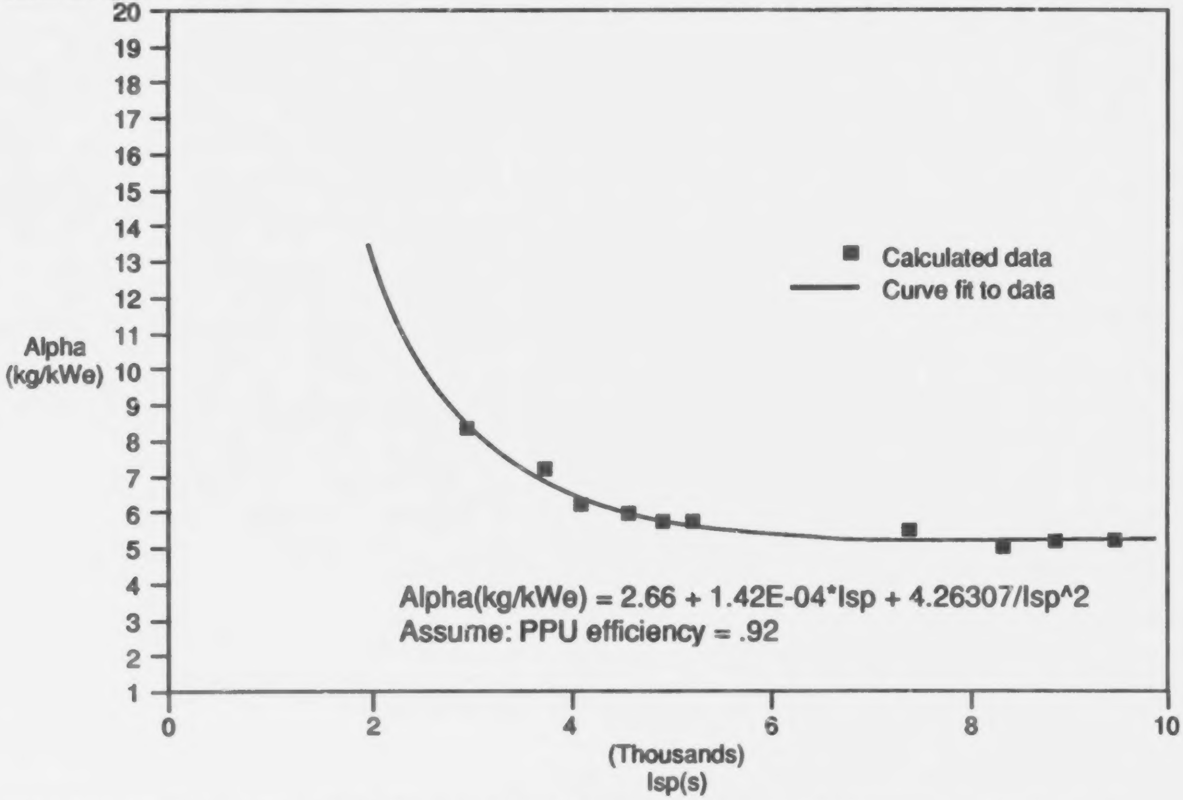


Figure 4.5-2.- Ion thruster system specific mass (includes single thruster, PPU, thermal, and structure masses).

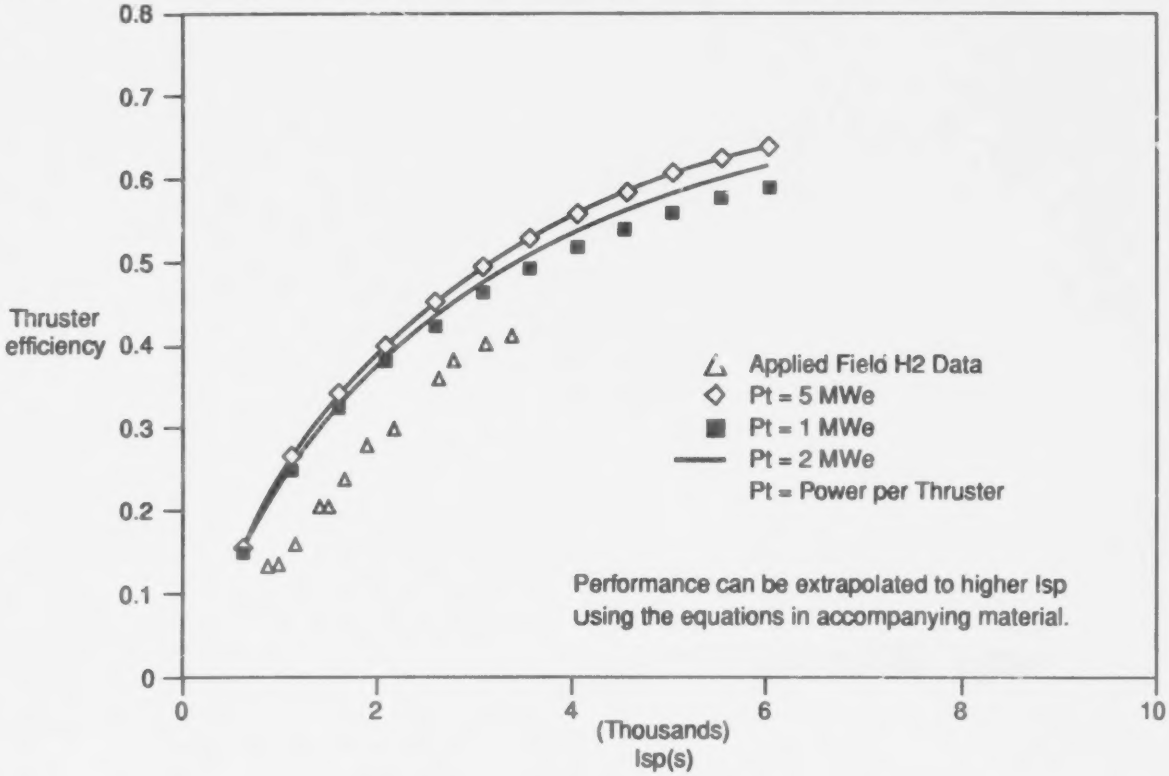


Figure 4.5-3.- Projected MPD thruster performance.

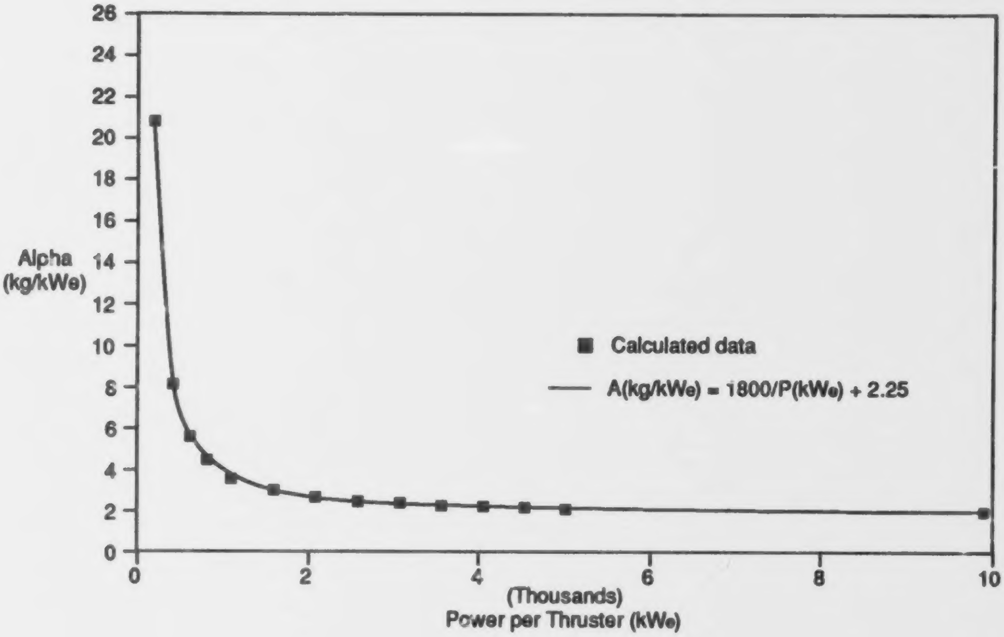


Figure 4.5-4.- MPD thruster system specific mass (includes single thruster, PPU, thermal, and structure masses).

Sp-100 Power for Nuclear Electric Propulsion

- Sp-100 type Liquid Metal-Cooled Reactor
- Advanced Technology Stirling Cycle Conversion

Stirling Cycle Parameters:	
Stirling Heater Temperature (°K)	1300
Sink Temperature (°K)	250
No. of Engine/2500 kWt	8
No. of Engine/No. of Operating Engines	1.14

Shielding Parameters:	
4-PI Shielding	
Separation Distance @A (meters)	40
Separation Distance @B (meters)	30

	DR (mrem/hr) (@A)	DR (mrem/hr) (@B)	Time (days)	Total Dose (rem) (@A)
Manned Vehicle	7	200	365	60
Unmanned Vehicle	42	200	5*	5

* Manned Proximity Operations of no more than 5 days

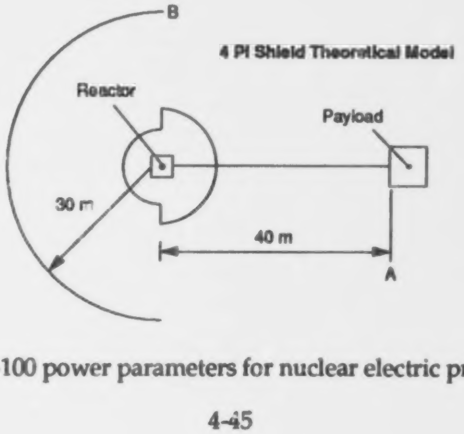


Figure 4.5-5.- SP-100 power parameters for nuclear electric propulsion.

TABLE 4.5-1.- SP-100 REACTOR SYSTEM DATA

				Manned		Unmanned	
Reactor Power (kWth)	Temp Ratio	Alternator Power (kWe)	Pwr Sys Mass (kg)	Shield Mass (kg)	Sys Sp Mass (kg/kWe)	Shield Mass (kg)	Sys Sp Mass (kg/kWe)
2500	2.0	593.8	9574	9137	31.5	3428	21.9
2500	2.1	632.6	10444	9137	31.0	3428	21.9
2500	2.2	667.9	11390	9137	30.7	3428	22.2
2500	2.3	700.2	12425	9137	30.8	3428	22.6
2500	2.4	729.7	13566	9137	31.1	3428	23.3
2500	2.5	756.9	14832	9137	31.7	3428	24.1
5000	2.0	1187.5	18944	10515	24.8	4141	19.4
5000	2.1	1265.2	20683	10515	24.7	4141	19.6
5000	2.2	1335.8	22576	10515	24.8	4141	20.0
5000	2.3	1400.3	24647	10515	25.1	4141	20.8
5000	2.4	1459.4	26928	10515	25.7	4141	21.3
5000	2.5	1513.8	29460	10515	26.4	4141	22.2
7500	2.0	1781.3	28352	11385	22.3	4601	18.5
7500	2.1	1897.8	30962	11385	22.3	4601	18.7
7500	2.2	2003.8	33801	11385	22.5	4601	19.2
7500	2.3	2100.5	36906	11385	23.0	4601	19.8
7500	2.4	2189.1	40328	11385	23.6	4601	20.5
7500	2.5	2270.7	44127	11385	24.4	4601	21.5
10000	2.0	2375.0	37748	12031	21.0	4949	18.0
10000	2.1	2530.4	41228	12031	21.0	4949	18.2
10000	2.2	2671.7	45014	12031	21.4	4949	18.7
10000	2.3	2800.6	49155	12031	21.8	4949	19.3
10000	2.4	2918.9	53717	12031	22.5	4949	20.1
10000	2.5	3027.6	58782	12031	23.4	4949	21.1

4.6 TELEOPERATED ROVERS IN SUPPORT OF HUMAN PLANETARY EXPLORATION

Study Overview. As has been considered elsewhere in this report, an Earth-controlled Mars Rover/Sample Return (MRSR) mission is an important option for the collection of data prerequisite to human presence on the martian surface. However, although rovers[1] can be an important scientific and operational tool on the lunar, martian, and martian satellite surfaces, they are not appropriate for all missions. This special study examines specifically the utility of rovers in Case Study 1, Human Expedition to Phobos, and addresses generally the applicability of rovers in future manned missions.

The first part of this study was the examination of the utility of martian surface rovers in Case Study 1 by a team assembled at the Jet Propulsion Laboratory (JPL). The rover scenario for this case study was divided into two phases: the search and mark phase, which occurs between the arrival of the cargo vehicle and crew arrival, and the sample collection phase, which takes place after the crew's arrival. During the search and mark phase (figure 4.6-1) after initial setup and autonomous landing, the rovers' traversals and identification and marking of possible samples are controlled from Earth. In the sample collection phase (figure 4.6-2) the crew controls the rovers to return to sites of marked samples, to acquire samples, and to return samples to the ascent vehicle for rendezvous with the orbiting vehicle carrying the crew. In brief, their study indicated that, due to the circumstances of the Phobos mission, martian surface rovers controlled by astronauts in martian orbit offer few advantages over an Earth-controlled mission. A brief summary of those results will be presented in this section. A full report giving further details appears in volume III of this document.

Subsequent evaluation at the Ames Research Center (ARC) indicates that the use of rovers in Case Study 2, Human Expeditions to Mars, and on the lunar surface may be beneficial to both crew safety and productivity.

The primary study objective was to examine the feasibility and utility of controlling Mars rovers from martian orbit. Study methodology included evaluation of operational scenarios, rover performance expectations, infrastructure and crew support requirements, and degree of autonomy of the rover and crew workstation. The intent was to explore advantages to martian rover operation gained from the proximity of an astronaut crew.

Methodology. The JPL team evaluated the baseline Case Study 1 rover and crew operations based largely on their experience with the Mars Rover Sample Return (MRSR) mission. The study plan included the following specific tasks:

a. Define mission operations timeline.

b. Develop rover operations strategy.

c. Develop rover operations timeline.

d. Evaluate technology requirements/options.

Background; Key Assumptions.

a. Phobos mission as defined 7/20/88

b. Continuous communications capability between the rover and operators on Earth or in orbit

c. Up to 8 crew hours available per day (two crewmembers available up to 4 hours per day with no interruptions) for operating the rovers, an estimate based on discussions with JSC personnel with manned operations experience

d. Automated workstations in the piloted orbital vehicle for rover teleoperation by the crew

e. Landing risk equal to that for MRSR

f. Mission scientific objectives similar to those for the proposed MRSR

g. A slightly less advanced level of autonomy than is planned for the MRSR mission, to enhance crew utility prospects and ensure a conservative level of technology development

Findings.

Rover Use on Phobos. The JPL systems analysis of the scenario leads to the conclusions:

a. Teleoperation from the Earth or from a Mars orbit as opposed to the semi-autonomy of MRSR does not result in a higher data yield. The science return from this mission will be roughly half that of the 235-sol MRSR mission. Supporting evidence will be found in volume III of this document.

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[1] The term "rover" will refer only to rovers of the MRSR class—those teleoperated from Earth with some level of autonomy.

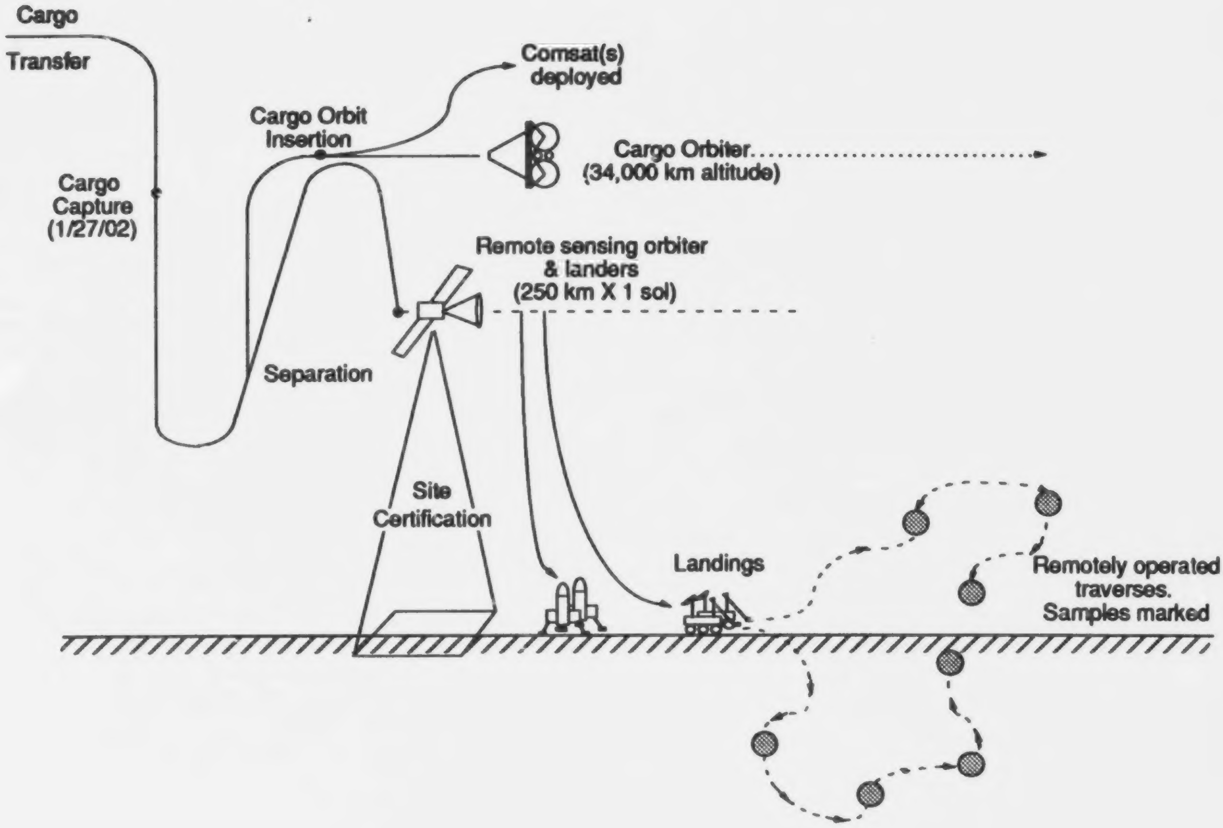

Cargo
Transfer
Comsat(s) deployed
Cargo Orbit Insertion
Cargo Orbiter (34,000 km altitude)
Cargo Capture (1/27/02)
Remote sensing orbiter & landers (250 km X 1 sol)
Separation
Site Certification
Landings
Remotely operated traverses. Samples marked

Figure 4.6-1.- Search and mark phase.

b. Teleoperation by a crew in martian orbit will allow relatively rapid response to anomalies. However, because of possible time-critical situations, expectation generation, execution monitoring, and reflex response are vital levels of automation required on any MRSR-class mission to reduce risk and enhance mission return.

c. The acquisition phase will allow little time for sample analysis beyond imaging of the acquired sample's surface, or for science feedback on the final sample selection. The time limitations could also cause minor hardware or scheduling problems to become catastrophic to mission goals. For example, because of the short crew time for teleoperation in Mars orbit, a hardware problem which required a week to circumvent would have a devastating effect on scientific productivity.

d. After months of relative inactivity, the crew will be placed in stressful situations requiring short bursts of exceptional performance and concentration.

e. The semiautonomy required for rover survival is relatively easy when compared to other technology needs for a human expedition.

In an MRSR-like mission, the key scientific parameters include the quality and number of samples returned. The Phobos expedition Mars rover does not meet MRSR objectives for sample returns by a factor of two or three. Although the crew's presence in orbit is useful for teleoperation, the hours available at Mars are insufficient to take advantage of that fact. Both nominal operations performance and rover contingency avoidance arguments suggest a need for a level of autonomy comparable to that planned for MRSR. A serious challenge which cannot be avoided is the need for a high degree of automation in the landing hazard avoidance for the rover/ascent vehicle lander. The semiautonomous rover capabilities suggested by these studies are technically challenging, yet probably achievable. In fact, at least three techniques for offroad semiautonomous traversal have already been identified and demonstrated to be feasible by JPL, FMC, and Carnegie-Mellon University (CMU).

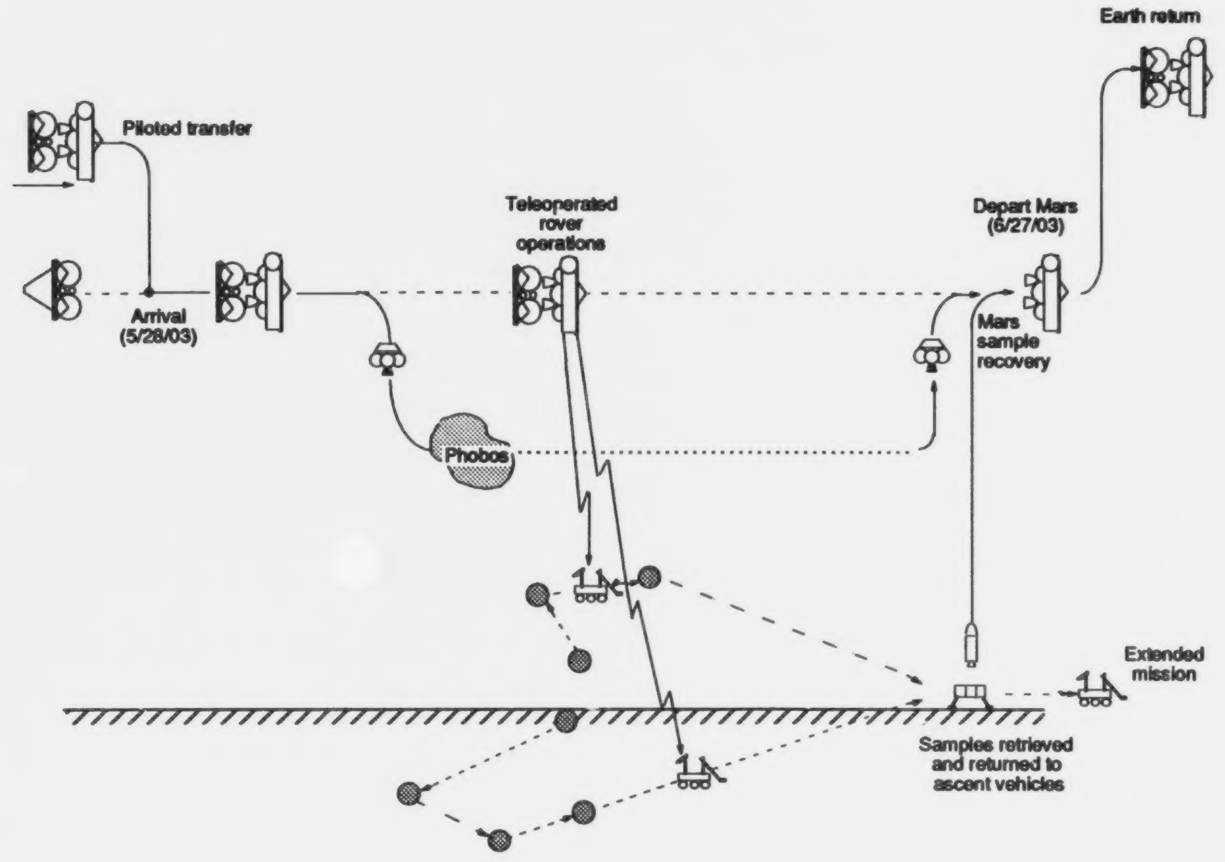


Figure 4.6-2.- Sample collection phase.

Rovers in Other Missions. The above negative evaluation of rover utility is in part a consequence of the special conditions imposed by Case Study 1. The results of this study might be different and the rovers might prove quite useful if the crew has an extended stay in Mars orbit or if the crew explores the same surface region the rover does.

The utility of rovers in other missions can be better understood by examining the use of similar devices on Earth. A good example is the use of aquatic rovers in scientific work in the waters below frozen lakes in Antarctica. Human divers do work in those waters; however, rovers are commonly used because they can safely scout large areas for long periods of time while the human operators remain in relative safety and comfort. Once an interesting location is found, the humans can investigate that site with a clear image of the environment to be encountered. The planetary surfaces are a dangerous and demanding environment much as the waters of Antarctica are, and rovers may play a similar role in facilitating safe exploration methods and extending mission duration capabilities.

In Case Study 1, orbiting astronauts are used as robust and sophisticated controllers of surface rovers. The above analysis indicates that astronauts in orbit may not add much to MRSR-like objectives because MRSR is designed to operate without close human presence. The advances expected for such a mission are in those areas requiring time-critical response at a level equal to or greater than that provided by humans.

On the other hand, rovers can be expected to add to the operational safety and effectiveness of astronauts when used to support them on a planetary surface. The immense utility and versatility of astronauts in scientific functions on planetary surfaces was not addressed in this study.

Planned FY 1989 Activities. Besides the utility of rovers as scientific instruments, they may also have utility as tools in easing the burden of planetary surface crews. Two examples are the use of rovers to locate appropriate landing sites and instrument them with navigation aids, and the use of rovers in various construction activities, an area now under study that will be explored in detail during the next fiscal year.

4.7 LUNAR OBSERVATORY STAYTIME EXTENSION STUDY

The objective of this study is to determine the mass of an advanced solar-based power system to enable a crew staytime extension into the lunar night during the construction phase of a lunar observatory.

Background. In the Lunar Observatory Case Study, the crew staytime on the lunar surface is limited to a lunar-day period (two Earth weeks) when sunlight is available. This scenario is predicated on the use of conventional energy storage options which are very massive, precluding the use of a solar-based power system for extending staytimes through the lunar night. However, technology advances in hydrogen-oxygen fuel cell technology could make energy storage for the long lunar night feasible.

A regenerative (rechargeable) fuel cell (RFC) could be used for multiple lunar night stays, given sufficient power generation during the day for recharging. A significant increase in construction capabilities is projected by extending the crew staytime, even if crew activity is reduced at night. The crew could resume full activity during the lunar day to complete the construction phase of the observatory and bring it online sooner than would otherwise be possible. In addition, the power system used for the construction phase could be integrated with or augment the power system for the operational phase of the mission.

Key Assumptions. RFC's were assumed for multiple lunar nighttime storage and for powering three vehicles with an activity sequence limited to 8 hours roving and 16 hours charging during lunar day periods only. Amorphous silicon photovoltaic arrays were sized sufficiently to supply the required power of 30 kWe for the habitat during the day and 15 kWe during the night, as well as enough power to sufficiently recharge three rovers designed to discharge at 13 kWe total. Only half the maximum power use (15 kWe) is anticipated during the lunar night due to the reduced activity of the crew. Gaseous storage of reactants was assumed for both primary and regenerative systems.

Findings. Initially, a primary fuel cell (PFC) (i. e., not rechargeable) was baselined to support a stay extension of only one lunar night. However, the preliminary analysis revealed that for an additional small mass increase, the system could be reconfigured to be rechargeable, thereby enabling continuous construction through multiple lunar nights.

A power system with RFC storage capable of supplying the required continuous day/night power is estimated to have a mass in the 7.0 to 8.0 metric ton range. The use of RFC's allows the construction crew to remain through multiple lunar nights. The resultant enhanced productivity of the crew may enable erection of the entire facility at one time. The detailed component mass estimates are shown in table 4.7-I.

Issues/Open Items. The crew staytime extension through lunar nights could have a significant impact on the total mass in low Earth orbit by eliminating one or possibly two launches.

The construction-phase power system could be designed for easy integration with the operational power system and be used for peak or contingency power needs.

Determination of the total impact on the mission requires formulation of a conceptual design and layout analysis of system requirements and subsystem interface requirements. Also, the impacts of crew time extension and integration of the construction power system with operational ones must be evaluated.

A primary fuel cell system designed for an extension of one lunar night would probably use cryogenic reactant storage, substantially reducing tank mass. Since cryogenic storage is not advantageous for the regenerative system, the primary fuel cell would have a greater mass advantage than the results indicate if one lunar night is the maximum extension contemplated. For more than one lunar night, however, the additional reactant mass for a primary system appears to be greater than the gaseous storage penalty for a regenerative system. These comparisons have not yet been quantified.

Planned or Required FY 1989 Activity. In FY 1989, a study should be made to trade the mass savings from eliminating one construction crew launch against the mass penalty for nighttime power storage. Another study could define mission impacts of integrating the construction power system into the operations power system.

TABLE 4.7-1.- POWER SYSTEM COMPONENT MASS ESTIMATES

Component	Mass (t)	Comments
Array	0.358	Area 770 m^2
Storage	4.0	Regenerative fuel cell (RFC) Primary fuel cell (PFC) (3.7 t)
Power mgt. & distribution	2.3	55 kg/kW (20-110 kg/kW user dependent)
Subtotal	6.65	
Vehicles		
Construction (2)	0.750	5.0 kW each (RFC)
Dune buggy (1)	0.225	3.0 kW (RFC)
Subtotal	0.975	
Total mass	7,625	

4.8 PHOBOS EXPLORATION ASSESSMENT

The objectives of this assessment, begun in June 1988, are to define the trade space to be used in succeeding study tasks, begin the definition of various study requirements, and define preliminary studies to be started in FY 1988 and future studies for FY 1989. A team was assembled and has begun to fulfill the objectives. Team membership comes from various NASA centers and has expertise in science, automation & robotics, transportation, operations, and mission analysis and system engineering (MASE).

The team has provided insight into the numerous areas needing study. Figure 4.8-1 describes the trade space needing study. As indicated, some items were considered in the short-term FY 1988 tasks and will become, with others, part of the FY 1989 tasks. Others may not become a part of future studies.

Case Study 1 defines the first human expedition to the Mars moon Phobos, which presents the first chance for humans to explore the surface of and assess exploration operations for an asteroid-type body. Phobos presents a unique set of environmental characteristics which must be considered in mission and contingency planning so that options can be preselected to ensure mission success. Two study tasks defined in FY 1988 were to provide preliminary information on anchoring to the surface in low gravity and on characteristics of flight over the surface. The results will help define vehicle and equipment requirements.

4.8.1 Surface Anchoring Methods

The objective of this preliminary study is to investigate and determine candidate methods of surface anchoring on Phobos.

Background. The small gravity force on Phobos would keep a motionless body on the surface. However, the potential is high for leaving the surface for extended periods of time (10 to 30 minutes) due to inadvertent pushes or bouncing off terrain features, indicating a need for anchoring methods to maintain surface contact and to create a stable work platform for sampling.

Key Assumptions. Assumptions include an extravehicular mobility unit (EMU)-clad crew and a modified manned maneuvering unit (MMU)/EMU vehicle/crew combination. The hardness of the surface is unknown; thus, assumptions for the parameter will be made. Anchoring entails physical contact with and attachment to the surface soil or rock. This study will not deal with the use of thrusters. Although tethers with single-end attachment points will not be considered, movement along tension lines or wires with end-point attachments to stable platforms/pins will be studied.

TRADE SPACE DEFINITION

PHOBOS/DEIMOS SURFACE EXPLORATION

What jobs?	Transit method?	Vehicles & Equipment?	Comm? GN&C?	Habitat?	Logistics & resupply?
Sample collection & packaging	▨ Orbiter flyaround	▨ Mars orbiter	Local/global comm	Whichever large vehicle is at Phobos	Self-contained but can be replenished from large vehicle at Phobos
Still photography	▨ Phobos Sortie vehicle flyaround	▨ Phobos Sortie vehicle	Local/global data transfer		
Movie film and Video	▨ EMU/MMU flyaround	▨ Enhanced MMU	Relative & global navigation	▨ Surface anchored and stationkeeping flight	
▨ Visit Multiple sites	▨ Surface anchoring	▨ EVA retriever	▨ Guidance & targeting algorithms		
Package emplacement	Rovers (Teleop/auto)	Rovers (Mars/Phobos)			
Visual observations	▨ Mountain climber EVA	▨ Enhanced EMU			
Sample inspection		▨ Anchors & penetrators			
Onboard sample analysis		Emplacement packages			
Map preparation		Photographic devices			
Mars observation		Sampling Tools			
Mars' rover teleoperation		In situ analysis tools			
		Dust protection			

▨ **Items considered in FY88 tasks.**

NOTE: This table presents elements to be considered as a part of each of the heading questions. No attempt has been made to identify horizontal correlations and none are implied.

Figure 4.8-1.- Trade space definition – Phobos/Deimos surface exploration.

Approach. The study approach consists of investigating potential anchoring systems based on Phobian surface characteristics and gravity force assumptions, sampling view/reach requirements, safety considerations, etc. This information will be obtained from existing reports, the science community, and other experts in applicable fields. Figure 4.8-2 shows an example of an anchoring device.

4.8.2 Flight over Phobos Surface

The objective of the second assessment task is to determine delta velocity requirements for flight activities on or near Phobos.

Background. The ease of traverse by flight over Phobos, the capacity to select areas for touchdown, and the ability to avoid obstacles are benefits that will be countered by the need to carry thruster propellant and to employ guidance, navigation, and control (GN&C) flight systems. The required velocity change of these flights will increase vehicle and system requirements when integrated with the scientific exploration objectives. The global exploration and operations called for in the scientific objectives will be limited by the current EMU/MMU systems.

Key Assumptions. The following assumptions are made:

a. Gravity potential models of Mars and Phobos will be available for the trajectory program.

b. A model of Phobos' physical size will be generated.

c. Flying vehicles will carry and use propellants, generate rocket plumes that may cause an airborne-dust problem, and require systems such as GN&C.

Approach. The study approach is to analyze the motions and delta V usage of various flight trajectories over the surface. These will include both short round-trip traverses of less than 2 km and long round-trip traverses of up to 10 km. Figure 4.8-3 shows some typical flight trajectories.

Planned FY 1989 Activity. Study requirements are still being gathered and FY 1989 tasks are still being defined. Additional studies will need to be incorporated to adequately cover the Case Study 2 and Case Study 4 activities. Work is continuing.

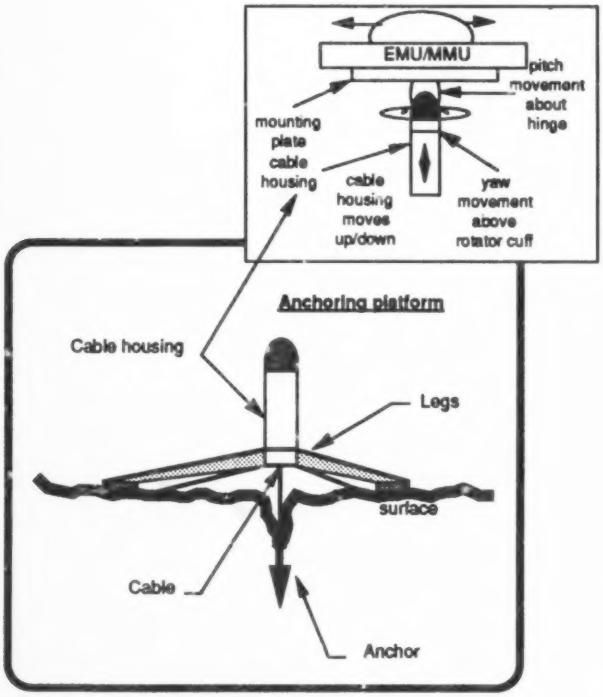
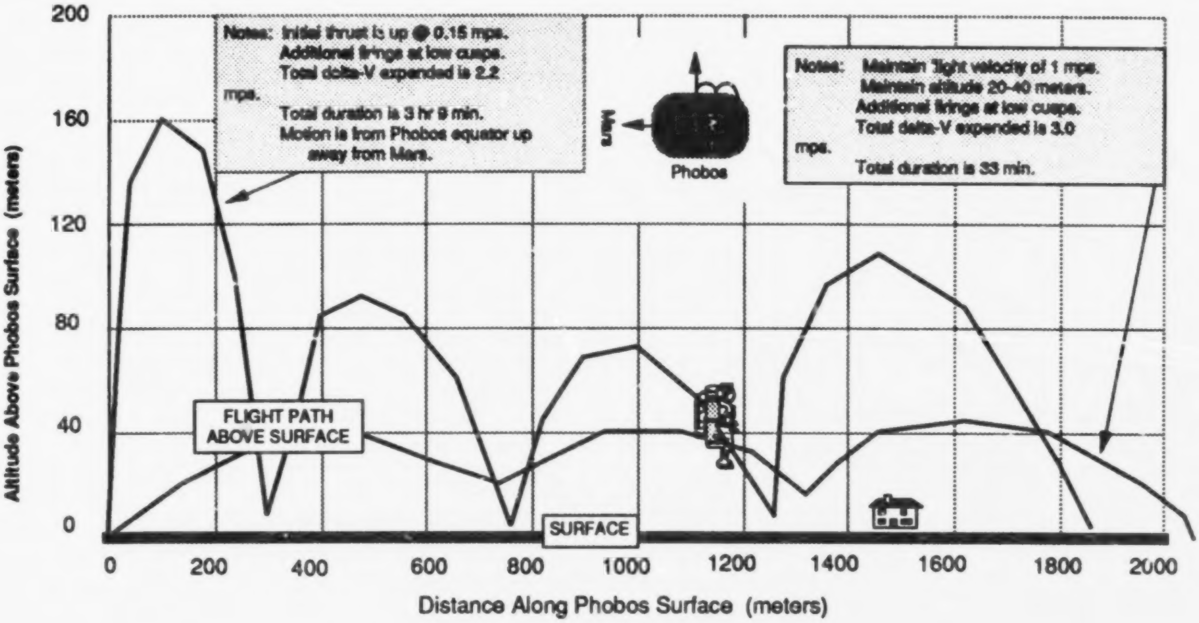


Figure 4.8-2.- Sample anchoring platform.



Notes: Initial thrust is up @ 0.15 mps.
Additional firings at low cusps.
Total delta-V expended is 2.2 mps.
Total duration is 3 hr 9 min.
Motion is from Phobos equator up away from Mars.

Notes: Maintain flight velocity of 1 mps.
Maintain altitude 20-40 meters.
Additional firings at low cusps.
Total delta-V expended is 3.0 mps.
Total duration is 33 min.

A.L.DuPont 07/18/88

Figure 4.8-3.- Flying over Phobos.

4.9 SPACE EXPLORATION COST UNDERSTANDING

The objective of this study is to conduct an indepth assessment of costing methodology to be applied to major initiatives still in the concept stage, with implementation far into the future where experience and techniques are very different. The task involves collecting experience from Agency-wide and industry-wide sources and fitting that experience to the environment of the initiatives. A key product, in addition to the costing techniques analysis, is a tailored costing method ("cookbook") for use by the human initiatives program agents that would include the programmatic and specific agency assumptions of the environment in which the initiatives will be developed. The methodology will be provided to each program agent for a distributed cost estimate of the program/scenario pieces. Other objectives are to develop a plan for changing the environment in which exploration programs are developed to make them more affordable, and to provide an interim capability to estimate relative costs and schedules for exploration case studies beginning in FY 1989.

Background. With no change in the way NASA does business, the budget for manned planetary exploration missions will have to be three or four times greater than the current budget. It is not likely that the American public will support such an expensive program. The choice is: change the way NASA does business or forget about going to the Moon or Mars.

Key Assumptions. In traditional cost estimating models, it is assumed that historical trends and methods of doing business will continue in the future. This is a major assumption that is not made in the current development effort. Instead, the analysis will attempt to understand the programmatic factors that drive cost.

Approach. The basic approach of this task is to develop a method to define conditions under which a certain cost outcome might result. The method must take into account not only the physical and technical characteristics of the program to be estimated, but also the programmatic environment in which it is to be developed.

Products. The following documentation is related to this study.

a. ECON, Inc. "Advanced Manned Missions Cost Model: First Progress Review," May 19, 1987.

b. ECON, Inc. "Advanced Manned Missions Cost Model: Second Progress Review." Viewgraph pres-

entation to NASA technical contract management, July 9-10, 1987.

c. ECON, Inc. "Advanced Manned Missions Cost Model: Third Progress Review" Viewgraph presentation to NASA technical contract management, Sep. 24, 1987.

d. ECON, Inc. "Advanced Manned Missions Cost Model: Fourth Progress Review." Viewgraph presentation to NASA technical contract management, Mar. 24, 1988.

e. ECON, Inc. "Advanced Missions Cost Model Workshop." NASA internal workshop, July 7, 1987.

f. Mandell, Humboldt C. "Productivity and Cost Influences on NASA Programs," Dec. 3, 1987.

g. Black, Ken U., Michael E. Hanna, and Kelley Cyr. "Documentation on Search for Innovative Cost Estimating Relationships." Unpublished JSC papers, Summer 1987.

h. Black, Ken U., Michael E. Hanna, and Kelley Cyr. "Space Cost Estimation Via Goal Programming" Unpublished JSC article, Summer 1987.

i. Black, Ken U., Michael E. Hanna, and Kelley Cyr. "Determining Cost Drivers for Large Scale Projects: A Study of Space Missions" Unpublished JSC article, Summer 1988.

j. Cyr, Kelley. "Office of Exploration Cost Understanding Special Assessment Cycle 1 Review." Presented to NASA code Z, Feb. 17, 1988.

k. Cyr, Kelley. "Cost Estimating Methods for Advanced Space Systems." Presented to the NASA Cost Estimating Symposium, April 19-20, 1988.

l. Cyr, Kelley. "Cost Estimating Methods for Advanced Space Systems." Presented to the 47th Annual Conference of the Society of Allied Weight Engineering, May 23-24, 1988. Revised Aug. 1, 1988.

m. Cyr, Kelley. "Space Exploration Cost Understanding Special Assessment." Presented to the NASA Code Z Program Review, June 15-17, 1988.

n. Cyr, Kelley. "Space Exploration Cost Understanding Special Assessment." Presented to the NASA Code Z Program Review, July 19-21, 1988.

Findings. The products resulting from this effort have led to a new approach to modeling the cost of major programs. With this new model has come a new understanding of the factors that drive cost. In addition to traditional technical factors such as size, weight, quantity, and performance, it has been found that management technique or culture is a significant cost driver. Since technical factors are usually determined by the mission requirements, the culture is a most important factor for potential cost reduction. NASA culture has changed in the last two decades; however, future changes can be made to make programs more affordable. Some of the recommended changes are:

a. Acquisition cost realism along with unit production cost as a significant design requirement

b. Planned product improvements and maximum use of proven components and subsystems (especially commercial items)

c. Presence of a continuous alternative

d. Short and stable schedules for development and production

e. Experienced, small staffs with clear command channels and limited reporting

f. Effective communication with users for cost/performance tradeoffs

g. Early development phase funding for production and support considerations

h. Use of mass production techniques as much as possible at every level of hardware (and software)

i. Technology intensification only at reasonable rates as determined by the recognized technology manager

j. Minimum functional complexity of individual hardware elements

k. Drive for commonality among hardware elements

l. Hardware elements designed with substantial performance margins

Issues/Open Items. Major cost reduction requires major cultural change. Any cultural change, good or bad, will be perceived as a threat and will meet major resistance. Therefore, any NASA cultural change will either be very slow or require forceful, dramatic intervention. A slow cultural change will not produce results soon enough to influence exploration planning, and coercive methods of change are not likely to work at NASA. Institutional strategies other than changing NASA from within must be examined. Several alternative institutional strategies have been identified:

a. Retain the current NASA/contractor/university structure. Implement NASA cultural changes within the existing institution.

b. Restructure the NASA organization.

c. Spin off operational programs from NASA. Focus NASA on research and development.

d. Establish an entirely new organization with alternative financing methods.

e. Create an incentive subsidy for private-sector space exploration. NASA develops technology only.

The recommendation of this study is that OEXP initiate a white paper study to assess needed cultural changes and alternative institutional strategies for space exploration.

Planned or Required FY 1989 Activity. In addition to continuing the work discussed above, the major new requirement for FY 1989 is to provide an interim capability to estimate relative costs and schedules for exploration case studies. Optional approaches are to accelerate development of the ECON model, develop an interim cost model, or use existing cost models.

It is recommended that Code Z use existing cost models and implement them with local cost personnel at integration agent centers for early FY 1989 needs. The application of funding to the ECON model should be accelerated to support cycle 2+ (FY 1989) cost estimating. NASA should also implement a standardized nomenclature for hardware elements along the lines of the Air Force equipment designation scheme (e.g., AIM-9P). Code Z should also implement an automated, centralized data base for scenario and hardware element data as soon as possible.

4.10 SCIENCE OPPORTUNITIES IN HUMAN EXPLORATION INITIATIVES

Introduction. As NASA is exploring potential programs that fulfill the space policy goal of extending human presence beyond Earth orbit, it is important to consider what that means in terms of advancing scientific knowledge. Historically, the Apollo program of human exploration of the Moon nucleated and coincided with a "golden age" of space science, which included unmanned exploration of the solar system with missions like Viking (Mars) and Voyager (Jupiter, Saturn, Uranus, Neptune), as well as plans for the "Great Observatories" such as the Hubble Space Telescope. The Skylab program, which evolved from Apollo, nucleated other later programs, including Spacelab life sciences and microgravity science. Science objectives, while they may not be the principal motivation for undertaking human expansion, will generate many of the most visible accomplishments as missions are carried out. To optimize the scientific accomplishments to be achieved within a program of human expansion, a science strategy should be developed from the beginning of that program.

The Office of Exploration (OEXP) has adopted a number of case studies for analysis, which generally involve missions to the Moon and Mars. These destinations for human exploration missions require technical capabilities that are achievable within the next 20-30 years. They also appear to offer the best combinations of exploration, science, and utilization among a suite of possible targets which includes various higher Earth orbits and locations (such as Earth-Moon libration points), and various asteroids that pass near the Earth's orbit. OEXP has focused on establishing concepts for how such missions would be carried out, examining their feasibility, and determining which of various possible sequences of exploration might be preferred. The requirements or opportunities for science in these case studies have been developed in an ad hoc manner during the past year based on inputs from individual scientists, a few workshops, and the literature. The science content has not been scrutinized by a science oversight function. Thus the work done this year does not constitute a science strategy, but has looked to incorporate some ideas on science objectives into an engineering analysis.

It is difficult to project where scientific inquiry will be 30 years in the future. To the extent that current space science programs can be projected to that period, one can project expected instrumental or exploration capability, but not what will be learned or understood by currently planned or desired experiments and instruments. Nevertheless, it is not too soon to begin laying the groundwork for science, as the perception of the science potential will be the basis for the design of many features of the developmental program that will lead to a robust and sustainable human expansion program.

This section provides an initial assessment of the science opportunities in human exploration missions. It is not directly tied to the case studies, but the application to the various case studies should be clear to the reader. It needs to be developed in greater depth by NASA and the scientific community, to provide guidance to OEXP and to ensure that appropriate weight is given to science in the development of a human exploration initiative.

Generalized Science and Exploration Objectives. Exploration of space, particularly the region of space between Earth and Mars, will have the following general objectives:

a. To study the origin, history, and current state of planetary bodies of the solar system and to understand their relation to the earth and the origin of the solar system

b. To seek evidence for the origin and evolution of living organisms through identification of environments in which life could have existed or through identification of physical or chemical remains

c. To conduct studies of the universe that can be uniquely or effectively undertaken using the new environments that would be accessible in the human exploration program

d. To utilize the newly accessible environments (e.g., high vacuum on the Moon or at a libration point; 1/3 or 1/6 g, or very low microgravity) in studies of importance to other fields of science

e. To understand the abilities and limitations of human beings for extended tours of duty in the new environments, particularly long-duration space flight beyond Earth orbit and on planetary surfaces

f. To establish the feasibility and utility of establishing permanent human outposts on the surface of other planets

Rationale for Science Opportunities in a Human Exploration Program. A principal rationale for undertaking scientific investigations as part of a human exploration program would be that science can be advanced faster, or more directly, or more positively, or less expensively by the direct involvement of human beings. This point is argued against by the growing capability of clever machines, which have conducted significant exploration and scientific investigation remotely, in places where human beings cannot physically go. However, Harrison Schmitt has argued that it is the experiential character of human interactions that is essential, the "consequence of the use of our senses, our minds and our motivations. It is in the spontaneous observation, integration, and interpretation of the total dynamic situation in which they are existing and in their calculated response that a person finds his role as explorer, poet, and human being."[1] These special capabilities of humans as observers, integrators, and interpreters provide the greatest leverage in tasks such as the exploration of new environments, searching for subtle or uncommon features, modifying experiments or studies based on new information, and, in conjunction with physical capabilities, maintaining and repairing mechanical devices. In addition, many observations made remotely will never be truly believed until confirmed by "hands-on" human observation.

For the exploration missions, three general types of experiments/investigations can be considered:

a. Investigations that uniquely use the capabilities of people functioning in the space environment to advance scientific knowledge. Such investigations include direct exploration, wherein the human intellect itself can contribute new observations and react to the unexpected. This includes situations in which the presence of people operating machines locally (teleoperation) is effective in rapidly advancing understanding of the environment. Also included are activities in which the capability to install, maintain, upgrade, or repair equipment or experiments makes new investigations possible or affordable.

b. Investigations that take advantage (on the margin) of the opportunity to emplace significant payloads (or people) in new places. For example, if launch and transfer capabilities for human crews to the Moon and Mars are developed, the same vehicles can be used to transfer major scientific payloads as well. Another example would be the scientific investigation of phenomena associated with the long stay of astronauts in space, required by other objectives of the missions. Some investigations might also be incorporated in part to provide productive alternate activities for low activity periods (i.e., during long flights).

c. Opportunities that come about as ancillary products of nonscientific activities. For example, lunar or Phobos mining activity could provide substantial new opportunities for geological investigations of those locations, or the development of a closed ecological life support system may provide opportunities in ecological or biospheric studies.

In addition to the direct science objectives or opportunities of the human exploration missions themselves, exploration of the Moon and Mars requires the best possible knowledge of the environment being explored: to gain familiarity with the environment and optimize the time of humans for directly productive activities, to plan their scientific investigations to optimize the chance for significant major discoveries, to properly equip the humans, and to provide for correct design and safe operation of the spacecraft systems. The precursor missions to the human exploration case studies have been addressed in section 3.2.

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[1] H. H. Schmitt (personal communication) from H. H. Schmitt and L. T. Silver, "Man and the Planets, a Summary of the First Fairchild Conference," August 7-8, 1975; California Institute of Technology (Unpublished report).

Opportunities in Lunar Exploration. Concepts for missions to the Moon have focused on the objectives of establishing scientific observatories and exploration base camps; developing capabilities to produce resources for life support, propellants, or other uses; and exploring the potential uses of the Moon as a testbed for the establishment of self-supporting human outposts on other planets. Options range from observatories that are human-tended and visited from time to time for exploration or experiment maintenance and repair to versatile bases that support a wide range of activities with a permanent or at least long-term staff.

The Apollo and its precursor missions (Ranger, Surveyor), augmented by ground-based telescopic studies, have provided the basic information needed to establish operational facilities on the Moon and to develop detailed plans for its further geological exploration.

Geological Exploration. Many unanswered questions remain about the origin and history of the Moon and its relation to the Earth among which some major ones are

a. Is there evidence for a lunar core and early internal magnetic field?

b. Did a completely molten surface region (magma ocean) exist and was it the mode of origin of the lunar crust?

c. Was there a period in lunar history in which the Moon (and Earth) were heavily bombarded by large basin-forming collisions (terminal cataclysm)?

d. Can a periodic change in the impact rate (as suggested for Earth) be demonstrated?

e. Can a detailed record of changes in solar flux be deduced from studies of implanted nuclei and their reaction products in exposed surfaces?

f. What is the bulk composition of the Moon and its relationship to Earth?

g. How is the origin of the Moon tied to the history of the Earth?

As can be seen from the questions, they are more detailed than those that could be asked before the Apollo missions. The answers will lead to more general conclusions about other solar system events and phenomena, and will thereby rapidly advance our general understanding of planetary formation, the origin and early history of the solar system, and the origin and history of the Earth. The solutions require substantially greater capability than that of Apollo missions to search out, characterize, and analyze lunar rocks and soil materials. For example, the solution to many problems related to the external bombardment history of the Moon will be found in true three-dimensional investigations of the lunar regolith, extending from bedrock to the surface. Identification and selection of samples for detailed analysis will depend on capabilities for iterative analyses and ability to discern subtleties of regolith structure. Many fundamental problems of lunar history will require collection and analysis of a few carefully selected and documented samples at a large number of places, requiring greater range of mobility than available previously. Onsite sample assays will significantly improve the selection of samples for more detailed analyses on Earth.

Geophysical observatories on the Moon will need to be operated at many locations for long periods of time to take advantage of natural phenomena, such as large impacts, to probe the lunar interior and determine its structural and compositional properties. As on Apollo, emplacement and maintenance or upgrading of scientific instruments will benefit from direct human activity.

Astronomical Observatory. The lunar surface offers an environment that makes it an attractive place to put major astronomical facilities. Environmental advantages include

a. High vacuum

b. Stable base (lack of internal lunar noise and surface vibration)

c. Extensive surface available (observatories requiring kilometer or larger scale emplacements are possible)

d. One-sixth-g; absence of structural problems associated with large structures on Earth (wind loading)

e. Slow rotation, allowing long observing times

f. Far side permanently shielded from Earth

These environmental benefits must offset problems associated with the greater costs of transporting and operating systems on the Moon, compared to the cost of using instruments of similar capability in Earth orbit. In general, both complicated operations and delivery of propellant to LEO equal to about six times the lunar facility mass are required to deliver a facility to the lunar surface. That is about twice the requirement to deliver a similar facility to geosynchronous Earth orbit (GEO) or to a Lagrangian (libration) point. However, in evolutionary scenarios in which lunar propellants are available, lunar surface, GEO, and Lagrangian points require similar amounts of LEO mass for the same mass of equipment.

Concepts for lunar astronomical observatories that have been advanced for early lunar outposts and have a reasonable pedigree (no general acceptance by scientific community is implied) include

a. Very low frequency radio array

b. Very large optical interferometer array (figure 4.10-1)

c. Moon-Earth radio interferometer (figure 4.10-2)

These and other possibilities have been described in a workshop report, Future Astronomical Observatories on the Moon (Burns and Mendell, 1988). Observatory characteristics are shown in tables 4.10-I through -III.

Other Science Opportunities. A number of areas currently are poorly explored, but need to be investigated by members of appropriate scientific communities. Some potential areas for consideration are:

a. Life Sciences:

  (1) "Geologic" studies to further explore early organic chemistry and the nature of interplanetary organic matter that has accreted to the Moon

  (2) Studies of behavior of living systems in steady 1/6 g (are these possible in man-tended mode?)

  (3) Ancillary studies of closed ecological life support system development

b. Physics:

  (1) Space plasma physics, utilizing the location of the Moon and the Earth's magnetotail and potential for active experiments

  (2) Gravity wave detection experiments

  (3) Neutrino detectors and other particle physics experiments requiring very low cosmic ray and secondaries background levels (probably requires advanced outpost capability and use of indigenous lunar materials)

  (4) Very long flight path mass spectrometers utilizing high vacuum and low magnetic field lunar properties

These and other potential areas of research opportunity should be more thoroughly explored as part of developing a set of requirements for either the lunar observatory or evolutionary lunar base case studies.

Phobos/Deimos Exploration. Phobos and Deimos are moons of Mars which become exploration objectives in strategies involving Mars system expeditions or in evolutionary scenarios leading to outposts on Mars. Both objects are enigmatic. Their relationships to one another, to Mars, and to other solar system objects (asteroids) are uncertain. The orbit of Phobos is decaying and Phobos will disaggregate to form a mini-asteroid belt around Mars in times that are short compared to the age of the solar system. This belt will subsequently decay with time. Most data on these moons have been obtained from the Earth and with the Viking mission to Mars. The Soviet Phobos mission will add significantly to the understanding of Phobos and Deimos in 1989.

The current understanding of Phobos/Deimos is meager, consisting of

a. Good images obtained by Viking (100-m class resolution) (figure 4.10-3)

b. Reasonable understanding of shape and gravity field and density (2.2-2.7 g/cm^3)

c. Spectroscopic evidence of surface composition similar to that of carbonaceous chondrites

d. Evidence (flat-bottomed impact craters) that a relatively hard interior is covered by a mantle of looser regolith

The USSR Phobos mission currently en route should provide:

a. Very high resolution images of the surface of Phobos

b. Surface compositional information, obtained by active remote sensing with a laser-mass spectrometer experiment

c. Some idea of subsurface structure obtained through radio wave sounding

d. Structural information obtained through seismic detector

Small bodies like Phobos and Deimos within the gravitational field of Mars may have suffered one or more collisions large enough to totally disaggregate them to form a debris ring, which would then reaggregate, forming an internal structure not directly related to an original structure. Images of Phobos indicate that internal structures (zones of weakness), expressed at the surface as sets of grooved terrain, are present (figure 4.10-3).

The spectral observations in hand suggest that Phobos and Deimos are similar in composition (at their surfaces)

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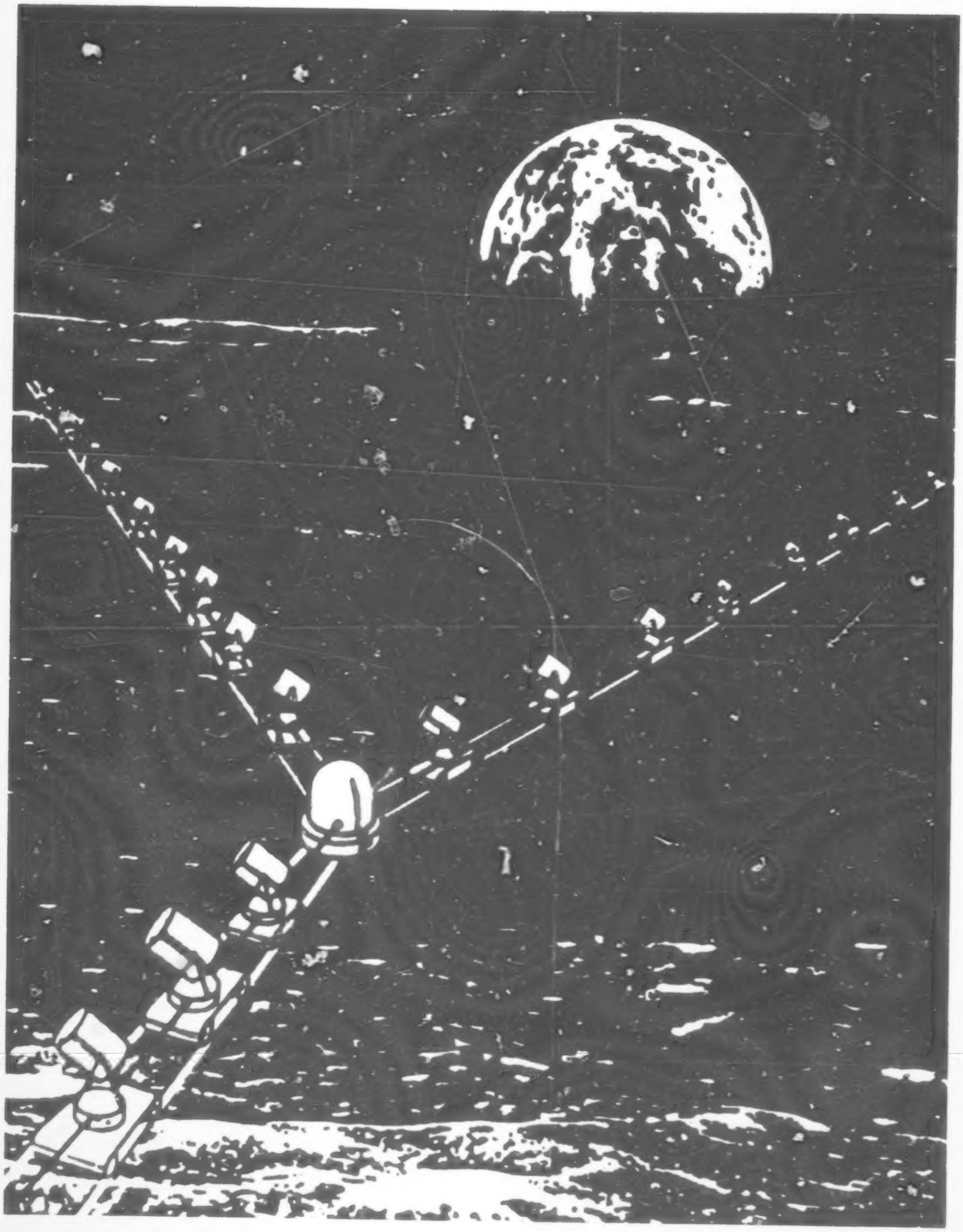

Figure 4.10-1.- Schematic view of an optical aperture-synthesis array on the Moon.

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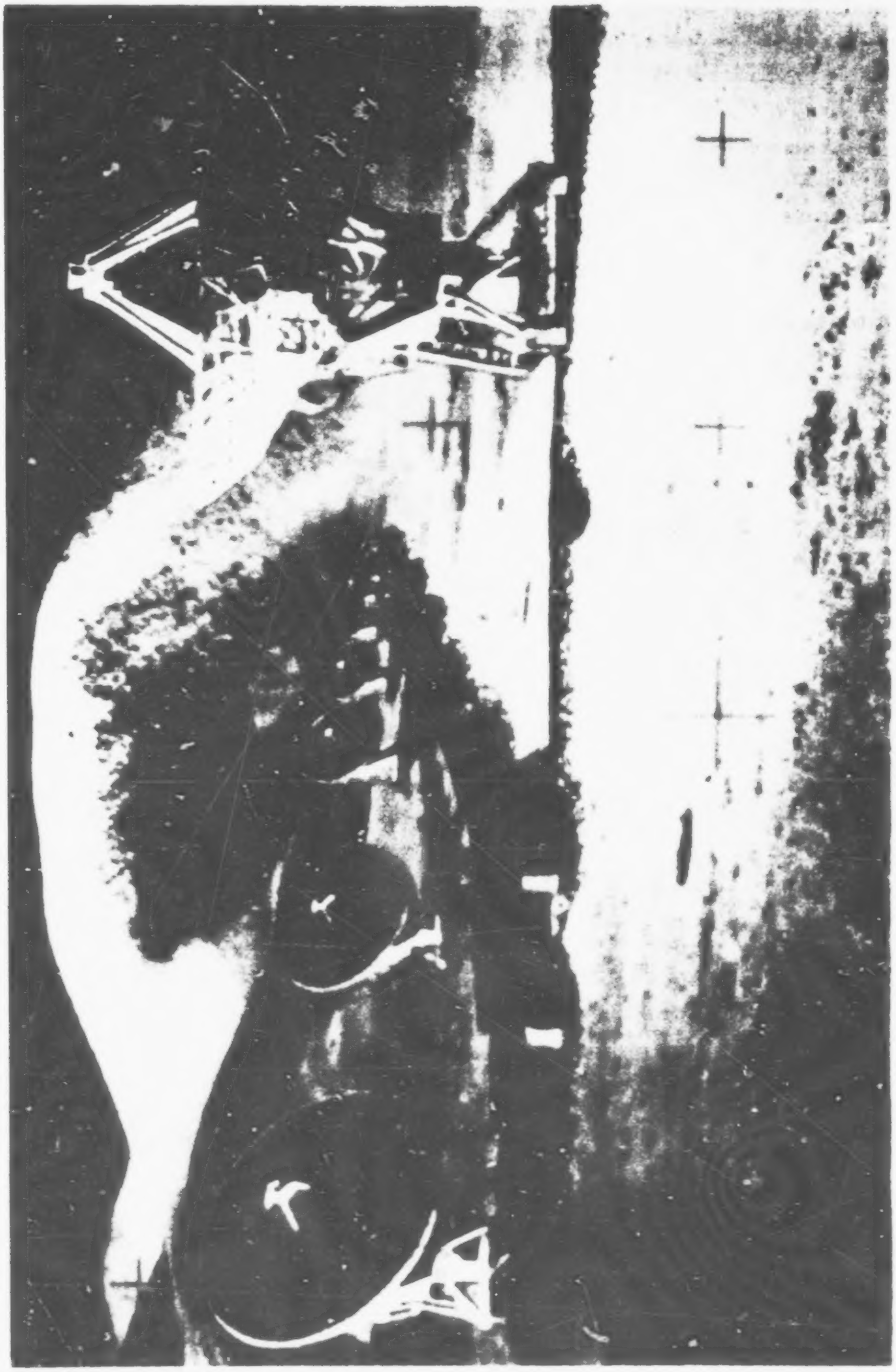

Figure 4.10-2.- Retouched photograph showing a very long aray of antennas constituting a radiofrequency interferometer emplaced on the Moon.

(From Burns and Mendell, NASA CP 2489).

TABLE 4.10-I.- VERY LARGE OPTICAL INTERFEROMETER ARRAY

(Concept proposed by Bernard Burke, MIT[1])

Experiment: 27 1-m optical telescopes in a y-array 6 km in length (figure 4.10-1).

Total telescope mass: approximately 7 metric tons (250 kg per unit), plus transporters and shielding.

Positioning requirement: control position and orientation of elements to 10 nm (100 angstroms) over 20 km separation.

Resolution and sensitivity: 10 microarcsecond resolution; light gathering capability equivalent to Palomar 5-m telescope. Several orders of magnitude increase over Hubble Space Telescope.

Operational modes: snapshots; subarrays doing different experiments; interspersed long observing sequence with short projects. Burke suggests, by analogy with Very Large Radio Array, that up to 1000 scientists might use the facility each year.

Significant research potential:

- Stellar analogs of solar cycle & sunspots could be studied on high resolution (100 pixel) images of stars in our galaxy.
- Characterize planets of nearby stars.
- Improve understanding of quasars, galactic nuclei, black holes through increased resolving power.

Rationale for establishing at lunar base: Uses stable base, allowing high positioning accuracy; requires human crews for emplacement and maintenance; structural mass may be lowered by innovative use of lunar shielding designs.

TABLE 4.10-II.- VERY LOW FREQUENCY RADIO ARRAY

(proposed by J. Douglas & H. Smith, University of Texas)

Experiment: Consists of a large number of small (50 g) simple dipoles established over a distance of tens to hundreds of kilometers (initial limited capability can be expanded by adding dipoles). Observes the long wavelength radio spectrum (300 m to 10 m).

Total telescope mass: Initially, perhaps a hundred dipoles with a mass on the order of 50 kg can provide significant new science.

Location: Far-side site is preferable to escape noise from Earth (ionosphere and manmade radio leakage).

Resolution: 1-0.1 degree resolution for galactic sources.

Significant research potential:

- First map of radio sources for wavelengths longer than 30 m and great improvement at wavelengths above 10 m.
- Observe nonthermal radiation from plasma instabilities (Sun, Jupiter, Saturn).
- Study galactic synchrotron radiation.
- Galactic plane studies.
- Previously unobserved features associated with galactic sources, compact nearby dust clouds, fine-scale structure in galactic emission.

Rationale for establishing at lunar base: Far side site is shielded from Earth's ionospheric and manmade radio noise; very extensive array (10's of km) requires stable base; human crews involved in installing dipoles and maintaining system.

Reference: Workshop on Very Low Frequency Radio Array (volume 3 of thisreport).

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[1] Burke, Bernard F. "Astronomical Interferometry on the Moon." In Lunar Bases and Space Activities of the 21st Century, ed. W.W. Mendell. Houston: Lunar and Planetary Institute, 1985.

TABLE 4.10-III.- MOON-EARTH RADIO INTERFEROMETRY
(proposed by J. Burns, University of New Mexico[1])

Experiment: two component radio array consisting of the space-based astro-array and a lunar very long array (VLA). The lunar VLA consists of up to 30 antennas (initially, only one) separated by distances from 200 m to 10's of kilometers (figure 4.10-2).

Total telescope mass: Depends on number of antennas. Individual antenna mass is to be determined, but would be perhaps 1 t.

Maximum instantaneous separation: 500,000 km (Earth-Moon).

Resolution and sensitivity: resolution at 10 gHz is 12.6 microarcseconds;resolution at 300 GHz is 0.4 microarcseconds; sensitivity and mapping capability increase with number of antennas on Moon and between Moon and Earth.

Significant research potential:

- Astrometric capabilities can provide improved celestial coordinate system, expansion of radio sources in extragalactic jets; extragalactic H_2O masers can be used as a new measure of Hubble parameter.
- Synthetic aperture measurements with high sensitivity to allow radio-burst mapping on other stars, mapping core of milky way galaxy, mapping accretion disks around compact object radio sources.

Rationale for establishment on the Moon: Takes advantage of the Earth-Moon separation to obtain high resolution data. Even a small radio antenna on the Moon, working in conjunction with Earth antennas, can perform useful studies.

[1] Burns, Jack O. "MERI: An Ultra-long-BaselineMoon Earth Radio Interferometer." In Future Astronomical Observatories on the Moon, eds.

to carbonaceous chondrites. Their density is compatible with that interpretation.

Thus, some questions about the origin and history of Phobos and Deimos that are currently outstanding include

a. What do these bodies represent? Are they residues of the material that accreted to form Mars, are they separately captured objects, or do they have of some more complicated history?

b. When did their material form, when did they attain their current orbital characteristics, and when (and how many times) have they been disaggregated?

c. What is their composition, and how has that composition changed due to internal or external causes through time?

It is not clear at this time just how many of these questions will be answered or how they will change with the USSR Phobos mission. It is likely that sample return is required for validation of the remote measurements, but no such mission is currently planned.

Due to Phobos/Deimos position with respect to Mars, if materials useful as rocket propellants are discovered to be plentiful and extractable on either, a significant leverage would be available by using these propellants in subsequent Mars exploration missions (manned or unmanned). Thus, a scientific objective of the exploration is to establish the characteristics of the material well enough to evaluate that potential.

The structural complexity and the apparently primitive nature of the surficial materials of Phobos/Deimos suggests that direct human exploration will be required to answer the main questions of origin and history. Relations between various materials may be subtle and interpretation may depend more on textural or structural relationships than on differences of composition. Important observations may be possible only at specific spots, chosen by prior onsite observation.

Phobos and Deimos are small objects in terms of other planetary satellites, but they are still extensive in terms of human exploration capabilities. The surface area of Phobos is on the order of 1500 km^2, and Deimos, 400 km^2. Techniques of exploration in their milli-g-or-less gravity fields will need to be developed.

Surface physical properties are not currently known well enough to plan operations that require establishing significant facilities on the surface (cohesion, bearing strength, etc.), but may be better understood after the USSR Phobos mission.

If use is to be made of Phobos/Deimos materials as propellants in later missions, samples will need to be returned to Earth so that specific extraction techniques can be designed. This sample return can be carried out robotically (soil samples are most likely sufficient) or as part of a human expedition.

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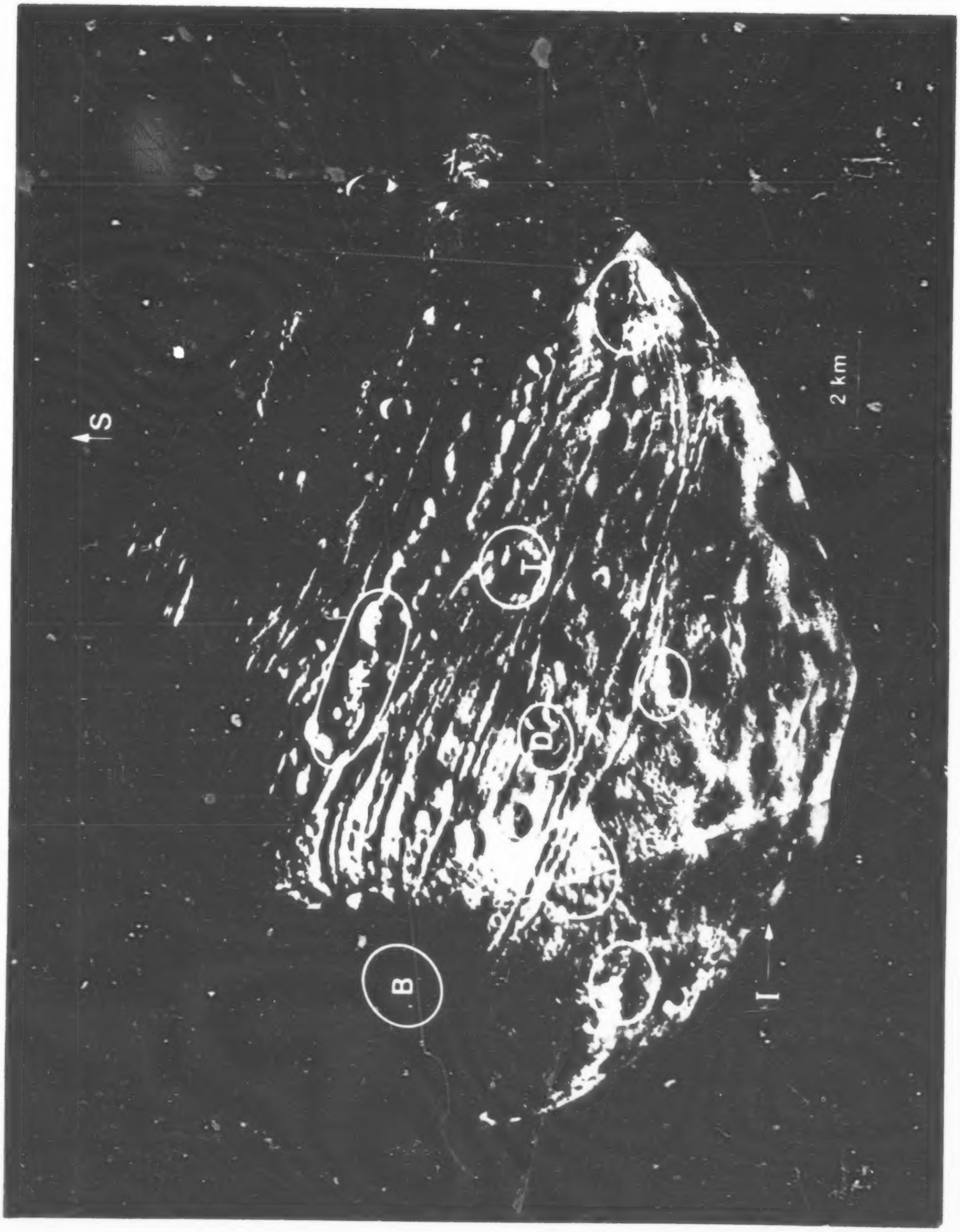


Figure 4.10-3.- Phobos observed at 880 km to reveal the presence of striations and crater chains.

Mars Exploration. Mars has been explored robotically by a series of U.S. missions (Mariner 4, 9; Viking 1, 2) and is the target for a series of missions planned or being discussed by the U.S. and U.S.S.R. between 1988 and the mid 1990's (Phobos, Mars Observer, Mars '94). These missions will provide a global view of Mars, closeup imagery, and experimental and analytical data at a few points on which further exploration of Mars will be based. Currently, the only missing element of the Mars exploration strategy is a significant commitment to returning soil and rock samples from Mars, which would provide calibration of the geological ages of materials, extend and certify the results of the remote sensing studies, and provide certification of the characteristics of the martian soil necessary to undertake human exploration safely. Such missions are in a definition stage in both the U.S. and U.S.S.R. programs.

The exploration of Mars in a reconnaissance mode can be carried out to a large extent by robotic devices. At the current state of understanding, it is difficult to speculate on the specific scientific objectives of a human expedition. Much will be learned in the next ten years which will focus the exploration objectives and hazards of human exploration of Mars.

In a general sense, there will be problems or issues identified by the robotic missions that will remain too complex or subtle to be resolved by robotic missions. The rate of progress in understanding Mars is partly controlled by the limited launch opportunities (every 26 months) and long mission times (3 years for sample return missions). A human expedition, allowing rapid iteration of studies involving sampling, sample inspection, and return to the sample locations for more detailed work, could significantly speed up the learning rate. It is conceivable that a properly equipped human expedition could shorten the time necessary to understand key elements of the history of Mars by decades.

A highly important science objective of Mars exploration is the search for evidence of existing (not considered likely) or ancient life. Conclusive evidence could possibly be obtained by robotic missions, in which case a more extensive exploration program would likely emerge. More probably, evidence is subtle enough and scarce enough that human exploration will be necessary to make the appropriate observations, interpretations, and inferences leading to the identification of specific environments or materials that can resolve the issues.

If humans are to be used effectively in such investigations, they must be appropriately equipped, both with tools and with knowledge gained from the robotic exploration. Therefore, an integrated Mars exploration strategy includes both the unmanned precursor missions and the human missions, to make the appropriate allocation of functions between the two types of capabilities.

The establishment of a permanent human outpost on Mars would require the utilization of indigenous materials and the environment to support the habitation. This requires an "operational" or applied science content for both precursor and human missions. Questions such as, where is water most accessible for life support and propellants, what are the detailed characteristics of martian soil and rock with respect to supporting the growth of food, and what are the surface and subsurface characteristics at proposed base sites, are in this category. Many of these could be provided by precursor missions, but will form some of the detailed science objectives of human missions as well.

Applied Life Sciences. Lunar and Mars exploration will stretch the capabilities of life sciences in the areas of

a. Long term health maintenance, medicine, health monitoring, and related areas

b. Psychological well-being and crew performance under conditions of isolation and long separations from Earth

c. Conditioning of humans to zero-gravity, fractional gravity, and artificial gravity, and readaptations when going from one level to another

d. Highly closed life support systems, including growth of food

e. Understanding of radiation biology and understanding and prediction of the space radiation environment and hazard to crews

f. Developing new capabilities for conducting human activities in new planetary environments (extravehicular activity capability, human-machine interface, autonomous operations, etc.)

These will form fertile ground for scientific advancement along with the development of new operational capabilities.

Provision should be made to conduct the appropriate experiments both in advance and as part of the human exploration missions and to interpret them broadly for use outside of the immediate area of application.

Accomplishments in 1988. The major accomplishments of this year have been associated with the first attempts to define a set of science opportunities for the four case studies described in this report. Details of the science complement for each case study can be found in other sections of the report.

Human Expedition to Phobos. The Phobos mission would accomplish the mission objectives of the first detailed geological exploration of a small body. Issues to be faced include the difficulty of operations in the milli-g field, and the need to optimize the expenditure of crewtime to obtain the greatest scientific benefit. A sample return is not required prior to the first manned Phobos expedition. However, in strategies that utilize Phobos propellants, a sample return mission can accelerate the development of a proven extraction process. Sample return would be a major element of the first human mission to Phobos, in any case.

Human Expeditions to Mars. The principal science opportunities for this mission are to explore the surface of Mars and to conduct exploration of Phobos and Deimos. Staytime is short and the capabilities of the crew will be limited. To make important new discoveries that could not have been made by relatively simple machines, an intensive automated precursor suite will be necessary to choose the optimum sites for making new discoveries and to provide the crew with enough information that they can perform their scientific tasks effectively.

Lunar Observatory. The potential for making major scientific advances through a series of "Grand Observatories" at a far side lunar base appears to be high. The observatory would be a multi-instrument facility; some instruments would extend over many tens to perhaps hundreds of kilometers from the base. They would offer operational benefits of access from a central location, but would only be serviced at intervals. Geological exploration of new lunar regions would be assured through repeated visits to the observatory and instrument sites, and through utilization of the system capability for landing at other important sites.

Lunar Outpost To Early Mars Evolution. The major objectives of this case are to develop the capability to produce propellants on the Moon to support the base and eventual Mars exploration, and to develop the life sciences and operational capabilities necessary for the Mars exploration. In particular, life sciences research on fractional-g adaptation would be an important element, utilizing the 1/6-g natural gravity and perhaps centrifuges to study effects at higher gravity fields. However, there is sufficient allowance in the science budget for the lunar segment to emplace the lunar observatory elements described in that case (although the lunar outpost would not necessarily be on the lunar far side). The objectives of the Mars exploration would be similar to those of the Mars expedition case, but the emphasis would be on developing long staytime capability. Thus, locating and developing capabilities to extract resources, particularly water on Mars and propellants on Phobos or Deimos, are more greatly emphasized.

Products. In addition to the case study evaluations, the following specific publications have been completed or are in work:

a. Jack O. Burns and Wendell W. Mendell, eds.(1988), "Future Astronomical Observatories on the Moon", NASA CP-2489.

b. Michael B. Duke and Donald A. Morrison, eds.(1988), "Precursor Missions to the Human Exploration of Mars, ZS-S-R-001, NASA Office of Exploration (See volume III of this report)

c. Mark J. Cintala, editor (1988), "Precursor Missions to a Lunar Base Program," ZS-S-R-002, NASA Office of Exploration (See volume III of this report).

Publications in preparation:

a. Jack O. Burns, editor (1989), Workshop on a Lunar Very Low Frequency Radio Array, NASA conference publication, in review (see volume III of this report).

b. G. Jeffrey Taylor, editor (1989), Lunar Geoscience from a Lunar Base, NASA special publication, in review.

c. Eugene M. Shoemaker, editor (1989), Human Exploration of Phobos and Deimos, in preparation.

Plans for 1989. Additional attention needs to be given to strategy and to the identification of specific experiments and facilities that should be incorporated into the case studies adopted by the OEXP. In particular, the allocation of science objectives to unmanned and manned missions in Mars exploration needs additional work; however, the relationship of possible observatories on the Moon to ongoing programs in the OSSA program need to be understood. More complete definitions of certain major experiments and facilities must also be understood, to ensure their feasibility from a transportation and operational point of view.

The activities that should be carried out in 1989 include:

a. Conduct additional survey workshops with elements of the science community to understand what the key science objectives or opportunities are for each exploration concept.

b. Begin the process of validation of science opportunities by conducting an integrated analysis of science objectives by a broadly based science community. This will be done initially through a workshop process, which can lead to analyses by more formally constituted bodies.

c. Create a task force of the NASA advisory council in FY 1989, will be created to advise the OEXP on goals, objectives and implementation planning. A subcommittee of that task force is envisioned, which will deal with the science issues, including strategy.

d. Seek validation of science objectives/opportunities and additional insights into the process of science strategy for the exploration missions from the Space Science Board of the National Academy of Sciences. This will be done at a later date.

4.11 POWER TECHNOLOGY WORKSHOP

A brief, first-of-a-series, power technology workshop was held at the Lewis Research Center (LeRC) on April 15, 1988. The workshop served as an open forum for Integration Agents (IA's), power technologists representing NASA Headquarters, LeRC, and other NASA centers, to freely exchange information, concerns, ideas, and issues. Representatives from NASA Headquarters' Office of Exploration (OEXP) and Office of Aeronautics and Space Technology (OAST) also participated in the workshop.

The first topic of discussion was the origin and rationale of the power requirements chart that appears in the Code Z Prerequisite Requirements Document (PRD). The ensuing dialog suggested amendments and augmentations that were made to the chart during the workshop (figure 4.11-1).

Each IA presented perspectives on projected requirements and issues. Projected power needs for nodes, spacecraft, and planetary surfaces were discussed. Concerns were raised about some of the preconceived attributes of the various power systems selected to satisfy the power requirements. Therefore,several action items were levied on LeRC to amend the power domain graph (table 4.11-I) and to develop a preliminary figure-of-merit list (table 4.11-II) for selected power systems.

An additional IA perception led to a quick-cycle study. The study analyzed the mass associated with adding advanced storage for the lunar night during the construction phase of the lunar observatory. The report of the study appears in section 4.7 of this document.

Lewis technologists presented overviews of the technological status of space power systems, encompassing the topics of nuclear power, isotopic dynamic systems, photovoltaics, solar dynamics, batteries, and fuel cells.

Future power workshops will be held as deemed appropriate.

4.12 LUNAR HELIUM-3 WORKSHOP RESULTS

The objective of this section is to provide information for assessing the feasibility, practicality, and advantage of mining helium-3 (He-3) from the lunar regolith to provide fusion power on Earth.

Background. Based on the work of the Fusion Technology Institute, University of Wisconsin, a workshop sponsored by the Office of Exploration (OEXP) was held April 25-26, 1988, in Cleveland, Ohio, to discuss the feasibility of mining He-3 from the lunar regolith for use in terrestrial fusion applications. Experts from the nuclear fusion, mining, and lunar communities participated. The workshop centered on two aspects: terrestrial fusion technology, specifically as it pertains to He-3 applications, and the technology required to mine He-3 from the lunar surface.

Approach. An overview of mining He-3 from the Moon and two applications concepts were presented at the opening session of the workshop. Two parallel working groups (Lunar Mining and Fusion Power) were then formed for indepth presentations and discussions. The final session included reports and findings of the parallel working groups, a panel discussion, and an open question-and-answer forum.

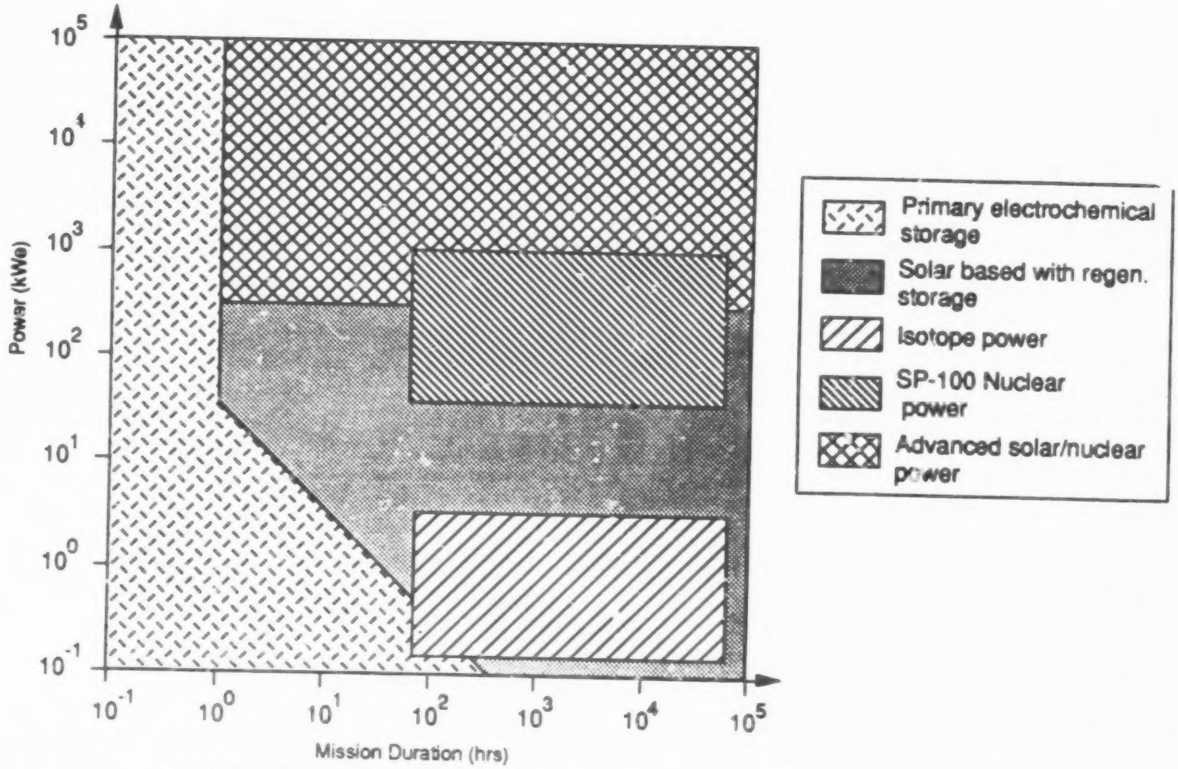
The fusion technology participants assessed the practicability of employing He-3 to advance terrestrial fusion power technology either as a main fuel or as a blanket material.

The space mission participants considered the viability of a lunar mining operation with respect to cost, enabling technologies, and timeframe.

The workshop as a whole considered whether the mining of He-3 for terrestrial fusion power applications could be a sufficient rationale for returning to the Moon.

Product. Workshop transactions were printed summarizing the working group discussions and panel report. The transactions also include charts, notes, and from invited speakers, rapporter and minority reports, and post-workshop papers submitted by attendees.

Findings. The Lunar Mining Working Group focused on several aspects of He-3 production on the Moon, including the actual mining operations, processing, transportation, infrastructure, and economics. The group concluded that mining, beneficiation, separation, and return to Earth of He-3 from the Moon are possible by means of a large-scale infrastructure and improvements in technology. Lunar oxygen production plants could provide an early technology demonstration (2010-2020) for He-3 production by demonstrating lunar soil mining and



Caution: This chart oversimplifies the power system arena. It should be used as a guide only and not a design tool. The actual power system design is heavily dependent on mission specific characteristics.

Fig. 4.11-1.

Figure 4.11-1.- Approximate power system domains.

TABLE 4.11-I.- EXPLORATION OF POWER SYSTEM DOMAINS

PRIMARY ELECTROCHEMICAL STORAGE

- Above about 1 hour and 50 kWe, the energy storage requirements favor regenerative systems. With regenerative systems all the energy need not be carried into space.
- At levels below 50 kWe, the energy requirements favor primary electrochemical systems as a low-weight, less complex system of choice.

SOLAR-BASED WITH REGENERATIVE STORAGE SYSTEMS (POWER GENERATOR COUPLED TO ENERGY STORAGE SUBSYSTEM)

- Beyond 500 kWe, civilian applications of solar-based regenerative power systems have not been currently identified.
- Below 1-hour missions, primary electrochemical systems are lighter due to low energy requirements.
- Above 10^5-hour missions, space environmental effects degrade performance of exposed components.

RADIOISOTOPES

- Beyond 5 kWe, the availability of isotopes becomes an issue.
- Below 100 W, primary electrochemical systems are lighter and less expensive.
- Below 100 hours usually means near-Sun missions, (high insolation), where solar-based regenerative systems are lighter.
- Above 10^5 hours, the half-life of isotopes becomes an issue.

SP-100 (SINGLE UNIT)

- Beyond 10^5 hours, fuel depletion for single SP-100 system limits applications.
- Below tens of hours, advanced solar-based regenerative systems may become lighter.
- 1000 kWe is the upper limit of power available from an SP-100 reactor. However, the SP-100 technology is not limited to this upper bound level.
- Below 50 kWe, reactor criticality factors result in higher specific weights. Solar-based regenerative systems are lighter.

ADVANCED SOLAR OR NUCLEAR POWER SYSTEMS

- Beyond the 0.5-to-1-MWe power level, both solar- and nuclear-based power systems have application. The choice depends on mission constraints such as weight, volume, area, complexity, reliability, duration, duty cycle.
- At mission durations of less than 1 hour, the choice of low-weight, low-cost power systems tends to favor primary electrochemical systems.

TABLE 4.11-II.- POWER SYSTEM FIGURES OF MERIT (FOM)

Technology	Mission	FOM units	State-of-the-art FOM	Potential FOM	Comments
PRIMARY ELECTROCHEMICAL STORAGE (SINGLE DISCHARGE)					
Lithium thermal battery	Munitions	W/kg	2000	4500	
		Wh/kg	12	120	
Lithium reserve battery	Various	W/kg	1000	4500	
		Wh/kg	200-300	300-400	
Alkaline primary fuel cell	Various	W/kg	100	8000	Power gen. portion
		Wh/kg	750	1200	Energy stor. portion
SOLAR-BASED SYSTEMS					
Solar dynamic/TES	LEO	W/kg	4.5	12	TES ?
		W/M^2	190	400	
PV-Si/Ni-H2 battery	LEO	W/kg	4.5	7	PV ?
		W/M^2	99	123	
PV-Si/Ni-H2 battery	GEO	W/kg	12	15	
PV-GaAs/NaS or lithium battery	LEO	W/kg	NA	16	
		W/M^2	NA	272	
PV-GaAs/NaS or lithium battery	GEO	W/kg	NA	28	
PV/RFC	Lunar surface base	W/kg	0.15	3	
PV/RFC	Mars surface base	W/kg	1.5	8	
ISOTOPE POWER SYSTEMS					
GPHS/TE	LEO to deep space	W/kg	5	6-7	
		Eff.%	5-10	10-13	
GPHS/Brayton	LEO to deep space	W/kg	6-7	6-7	
		Eff.%	26	26	
GPHS/Stirling	LEO to deep space	W/kg	NA	10	
		Eff.%	NA	30	
NUCLEAR POWER SYSTEMS					
SP-100 with TE's (Baseline)	Space & surface	kg/kW	NA	30	100-300 kWe
SP-100 with adv. dynamic tech.	Space & surface	kg/kW	NA	12	1000 kWe
MMWe reactor with Rankine or TE's	NEP	kg/kW	NA	7	

CAUTION: These data should be used only as a gross guide and not a design tool. The power system design is heavily dependent on mission-specific characteristics such as duration, environment, duty cycle, reliability, cost, and safety.

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processing techniques and by providing an opportunity to produce some He-3 as a by-product of the lunar oxygen production process. This is in keeping with the estimated timeframe when deuterium/helium-3 (D/He-3) fusion could possibly be ready for commercial terrestrial energy production (circa 2015).

The Fusion Power Working Group concluded that D/He-3 fusion offers significant advantages over fusion of deuterium/tritium (D/T): principally increased reactor life, reduced radioactive wastes, and high efficiency conversion to electricity. Probable greater public acceptability and reduced licensing requirements were also noted. There was a consensus within the group that a detailed assessment of the potential of the D/He-3 fuel cycle requires more information. A discussion of how to obtain this information engendered several ideas and a debate on which idea was the most feasible, rational, and expeditious. Finally, the group considered that while D/T fusion has received more attention as the mainline fuel cycle and may be closer to technical feasibility, D/He-3 fusion may be better for commercial applications.

Several recommendations to NASA were made on the final day of the workshop:

a. NASA should study the mining and return to Earth of lunar He-3 as an exploration program option.

b. NASA should join with the Department of Energy (DOE) to assess the potential for D/He-3 fusion and plan follow-on activities.

c. If D/He-3 fusion is determined to be a feasible option for terrestrial power production, a cooperative program should be developed between NASA and DOE to return lunar He-3 to Earth for terrestrial fusion power.

He-3 fusion power could have tremendous political, economic, and social impact. The possibility exists that abundant energy in the form of D/He-3 fusion could be made generally available worldwide at relative costs similar to, and perhaps less than, current energy costs. Perhaps of greater significance is the reduction and eventual replacement of fossil fuels as an energy source. Burning petroleum and high-sulfur coals contributes to acid rain and the greenhouse effect, vastly altering the environment and world climate, respectively. When a clean energy source is provided, these detrimental environmental effects can be reduced or eliminated, while rapidly disappearing reserves of petrochemicals can be retained for important applications such as medicines, plastics, lubricants, synthetics, and other chemical applications.

The NASA Lunar Helium-3/Fusion Power Workshop was a special emphasis study that does not directly affect any of the four cycle-2 case studies. The value of the workshop is derived from the possible impact it could have on future case studies. Not only could He-3 fusion power impact terrestrial energy production, there could be implications to space power and propulsion as well. Future case studies could include lunar bases with He-3 mining facilities and missions to Mars using D/He-3 fusion propulsion. Likewise, future case studies might emphasize the commercial applications of He-3 for both terrestrial and space energy production.

Planned or Required FY89 Activity. It is recommended that a joint NASA/DOE Lunar Helium-3 Fusion Conference take place in late spring or early summer 1989. Topics that should be covered include: physics requirements for D/He-3 fusion, mining and transportation techniques, terrestrial applications and commercialization of fusion power, space power, propulsion and environmental, legal, economic, political, and international issues.

It is also recommended that mining the Moon for He-3 be included in future OEXP case studies.

4.13 ADVANCED SPACE PROPULSION WORKSHOP RESULTS

The objective of this workshop is to provide a forum for discussion of mission analysis results available for assessing propulsion technology required to perform the lunar and Mars missions being studied by the Office of Exploration (OEXP). Another goal is to provide an opportunity for the cognizant community to consider advanced propulsion concepts and the adequacy of the analytic tools available to study the applications of propulsion technology.

Background. Over the past 25 years or more, there have been many studies of lunar and Mars missions using chemical propulsion, nuclear thermal propulsion, and electric propulsion devices. These studies have been based on differing ground rules and assumptions. The workshop described here was conducted as a forum in which study results could be presented to a cognizant group of NASA, industry, and academic representatives who are generally familiar with these studies. In addition to these studies, presentations were made on advanced propulsion technologies which included nuclear thermal propulsion such as nuclear energy for rocket vehicle application (Nerva) and more advanced systems, as well as rail guns, solar sails, etc. The participants were aware that due to differing ground rules and assumptions, it is very difficult if not impossible to quantitatively compare one propulsion technology with another. As technology decision points are approached, studies will be required to rationally choose the proper options.

Approach. Presentations were made on the gamut of pertinent topics by representatives from LeRC, MSFC, JPL, SAIC, SVERDRUP, Eagle Engineering, Astronautics Corp. of America, General Dynamics Space Systems Division, and the University of Texas. A detailed agenda for these presentations and workshop handouts are published in volume III of this report. These presentations provided a springboard for extensive discussion of topics briefly discussed in the following sections.

Products. The products of the workshop were charts provided to OEXP and explanatory text (see volume III of this document).

Findings. A synopsis of the workshop findings follows.

Trajectory Analysis. In general, the trajectory analysis techniques described in the presentations were adequate for the purposes of the studies. Concern was expressed that for studies of greater depth, the currently used techniques could lead to incorrect conclusions or impossible mission scenarios. For instance, launching to Mars from a space station is complicated by the nodal progression of the space station orbit. Simple trajectory scenarios are not appropriate for more than a broad-brush study. Launching to Mars from other transportation nodes does not appear to be well understood yet, either. The participants expressed concern that care be taken to ensure that trajectory sophistication is adequate to support study conclusions and scenarios.

Other trajectory characteristics implications affect operational issues, such as launch opportunities and launch windows. These need to be understood, as they could affect the choice of mission scenario or propulsion technology. For instance, the probability that low-thrust options have longer opportunities than do high-thrust should be investigated and understood.

Analytic Tools. The participants agreed that better mission and trajectory analysis tools will be required to explore the many options available for lunar and Mars missions. There are many tools currently available but most are old and were developed for computers two or more generations old. A systematic effort should be made to identify existing tools and their provenances. These tools should be evaluated and recommendations should be made on new tools needed and their sources of development.

Aerobrakes. Aerobraking at Earth and Mars is included in many of the studies currently concerned with transportation for lunar and Mars missions. Discussion seemed to establish that many of the assumptions associated with aerobrakes were based on return from geostationary orbit and were probably not adequate for realistic scenario assessment beyond the general concept level. In many cases, aerobrakes are assumed to be compatible with nuclear systems, artificial gravity, and practical launch and assembly. These concerns should be addressed in sufficient depth to permit realistic comparison with other propulsion options.

Nuclear Thermal Propulsion. Some of the studies presented suggest that nuclear thermal propulsion could be a strong competitor for lunar and Mars missions, particularly should aerobrakes not be as capable as has been assumed. NASA has considerable experience with nuclear thermal propulsion, which should be carefully considered, meeting head-on the safety, emotional, and political questions and problems inherent in the nuclear option.

Space Nuclear Power Operational Issues. A broader issue than nuclear thermal propulsion is the issue of nuclear systems in general—power as well as propulsion. The compatibility of nuclear systems with manned systems should be well understood before proceeding much further with mission scenarios. Nuclear systems may be enabling for lunar and Mars missions, and any limitations on their use, or the steps required to make them usable, should be identified in sufficient time to impact trade studies.

High-Power Electric Propulsion. The potential of high power electric propulsion systems, magnetoplasma-dynamic (MPD) and ion, remains to be demonstrated over the parameter ranges of interest. High power operation and thruster lifetimes are key issues. These questions should be laid to rest in time to either appropriately include or exclude consideration of these systems in comparison with other options.

Figures of Merit. The figures of merit for comparing the various scenarios for lunar and Mars missions are numerous and complicated in their interaction. Briefly, the potential figures of merit include cost, mass in low Earth orbit, mass ratio, human factors and environments, safety, trip time/stay time, modularity, packaging, launch opportunity, reusability, etc. If and when choices must be made between systems, rational choices dictate that these complicated interactions be considered. This topic should be considered before or at least concurrent with the conduct of mission studies to provide the data required to make selections. The complexity of the subject suggests that a workshop be held to initiate consideration of this tough issue.

Trade Studies. The participants identified a wide range of appropriate trade studies such as storable versus cryopropellants for ascent/descent propulsion, multimode missions, aerobraking versus propulsive options, and in situ propellant usage. A common and consistent set of ground rules was identified as a

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prerequisite to conducting studies to ensure the ability to compare propulsion options meaningfully and credibly.

4.14 ROBOTICS WORKSHOP RESULTS

An Emerging Technology. During the last couple of decades there has been tremendous progress in our ability to enhance the capability for humans to perform mental and mechanical tasks. Unfortunately, because this technology is nascent, is evolving extremely rapidly, and largely lacks analytical models, it is extremely difficult to make accurate predictions for the technology a decade hence. Two areas in which fairly accurate predictions can be made are commercially available computer hardware and classical control technologies; however, this predictability of realistic advances is rare in A&R technology. Although there are clearly directions in which work is being pursued with the anticipation of advancements, historically there is little evidence of accurately predicting the direction or magnitude of advances over the period of a decade.

However, in planning future manned missions to the Moon and Mars, it is important to make judgments concerning what is and what is not a feasible technology for one to two decades from today. For example: how realistic is it to assume that a particular in-space assembly task will be possible in the year 2002? To get a broader assessment, an Office of Exploration (OEXP) robotics workshop was held May 10-11, 1988, in Palo Alto, California.

Workshop objectives were to assess mission needs, current technology's adequacy, and the feasibility of needed advancements. Specifically, the workshop was designed to

a. Examine robotics requirements for OEXP missions.

b. Check the feasibility of current mission plans.

c. Identify system options.

d. Communicate the OEXP objectives and mission requirements to the university community.

e. Identify barriers and opportunities for missions.

Methodology. The workshop was attended by leading roboticists and NASA and industrial experts. It began with an overview of the OEXP program, the case studies, and the barrier issues being addressed, specifically in-space assembly. The attendees then split into five subgroups: navigation, mobility, manipulation, sensing and perception, and integration. These are interrelated but separate components contributing to the robotics requirements. Each subgroup was tasked to focus the problem by addressing such questions as: Where is this aspect of robotics applicable in future manned missions? What is the state-of-the-art today? Where will it likely be 2 years, 5 years, and 10 years hence assuming the current direction is followed? What areas require considerable effort before reaching a useful state? What are the barrier issues to the OEXP missions? What areas are potentially high leverage issues? What are some of the broad trades within these subcomponents? Through these questions the subgroups were able to provide the scope of the problem as well as identify the barrier issues, both of which are essential first steps in the definition and assessment phase. The workshop concluded with each subgroup presenting trans-scenario issues for its particular component of robotics.

Background. The overview of OEXP goals, case studies, and identified barrier issues was provided as a backdrop for the workshop's robotics assessments.

Findings. Some of the broad conclusions of the workshop follow.

a. The in-space assembly process is possible with reasonable extensions of current engineering and robotics technology, depending on appropriate design and structure (i.e., size) of the components. There was a strong feeling that with appropriate care taken in the design of components and the assembly process, the task is largely possible with robotic technology. One cannot separate the task, the component design and the design of effecting agent. It is unreasonable to assume that robots in space will take over all the assembly tasks currently done on Shuttles at KSC. However, robots may ease the assembly burden for well-designed processes.

b. All manipulated parts should be robot-friendly. For example, all parts should have clear machine-readable labels and be designed for easy robot grasping. Robot-friendly and EVA-friendly designs should be compatible.

c. Technology options for the OEXP missions were presented, such as design options for movement during in-space assembly. Movement is possible by flying and docking or by maintaining a structural attachment. Flying and docking options include thrusters, springers and hoppers, and grapple and reel. If mobility is achieved while maintaining structural attachment, design options include prehensile locomotion, rails and trolleys, booms and cranes, and serpentine locomotion. These options for robotic mobility will improve different design specifications on the pieces to be assembled, and will guide reasonable options for piece design.

d. Communications with the university community were opened.

A more complete report on results, including barriers, opportunities, options, and enabling technologies will be found in volume III of this report.

4.15 MINIMUM CREW SIZE FOR PHOBOS AND MARS MISSIONS

The objective of this section is to define the primary issues that will determine the number of crewmembers required for a Phobos or Mars mission. A representative example of the mass and operational impacts of a two-crewmember Phobos mission versus the four-crewmember baseline is also discussed.

The number of crewmembers sent on any manned space flight mission is determined by a balance of acceptable costs, the mission objectives to be accomplished, the mix of people and machines needed to accomplish them, and the acceptable level of risk to crew safety, mission objectives, and equipment.

Cost/Mass. The mass of crew and equipment is often taken as the leading parameter in determining the overall costs of a particular mission, as mass is the easiest parameter to quantify[1]. In turn, mass is logarithmically proportional to the volume of the spacecraft[2]. In general, as spacecraft volume and mass increase, costs will increase proportionally. Increases in the number of crew will increase mass of life support equipment and consumables and volume of required space for each crewmember.

Minimizing the number of crew may dilute this correlation between mass and cost. Minimum crew will cause much heavier emphasis on automation and robotics, with associated development time and costs that may not directly equate to the additional mass that will be incurred. In other words, a particular mission option may indicate a decrease in mass, but actually have an increase in overall cost due to expensive technology development.

The primary expense is incurred in getting the crew and life support equipment into low Earth orbit. Once that is accomplished, it is relatively inexpensive to support the crew onorbit and beyond. Therefore, it is expected that the cost-per-crew-hour in space will go down significantly when space flight evolves into longer duration missions. An additional cost parameter is the number of ground support people required. Fewer crewmembers onorbit and in-space may greatly multiply the ground support required to perform many of the functions remotely that are probably much more easily handled locally.

Functional Capability. Mission objectives are met by successfully accomplishing all of the mission functions sequentially by humans and machines. Limiting crew size limits the number of functions that can be accomplished successfully, limits performance, or increases reliance on machines to accomplish the equivalent functions. To quantify this impact, it is necessary to do a functional decomposition of the entire mission, then determine the amount of crewtime necessary to perform each function, crew endurance limits for each function, and the sequence of functions.

Effective crew mix is also significantly limited with fewer crewmembers. The remaining crewmembers on a scaled-down mission will be expected to have all of the discipline skills that existed in the larger crew. Therefore, each will probably be less capable in a greater number of skills. For example, the diverse skills of commander, physician, scientist, and engineer might have to be compressed into two people.

Human/Machine Productivity. In some cases it is possible to replace human productivity with machines. Unfortunately, such applications in the near-term are fairly limited. Machines best perform well understood routine tasks. Humans complement this function, preferring new innovative tasks that are initially poorly understood. Examples are plentiful. It has almost become routine for Shuttle flight crews and ground controllers to devise innovative fixes for unanticipated contingencies. Furthermore, the shuttle environment is well known, with procedures practiced hundreds of times. In the relatively unknown environment and situations that will be encountered in a Phobos or Mars mission, crew contributions will be even more critical. Considerable advances will be required in artificial intelligence, automation, and robotics before it is truly viable to replace crew members with machines.

Risk Assessment. Several factors must be evaluated for risk potential.

Human Psychology. Extensive studies have been done on the psychological impacts of small groups in confined spaces for extended periods of time. Fewer crew-members

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[1] Cyr, Kelly J. "Cost Estimating Methods for Advanced Space Systems." Presented at 47th Annual Conference of the Society of Allied Weight Engineering, Inc. Plymouth, Michigan, May 24, 1988. Proceedings to be published.

[2] Heineman, Willie, Jr. "Fundamental Techniques of Weight Estimating and Forecasting for Advanced Manned Spacecraft and Space Stations." NASA TND-6349, May 1971.

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decreases the options and diversity of human interaction. Unless psychological factors are well understood and accounted for, the success of the mission could be significantly affected.

Contingency Flexibility. This includes the ability to meet mission objectives (fail-ops capability), to ensure crew safety (fail-safe capability), and to prevent undue equipment wear and damage when events don't go according to plan. As stated previously, humans perform well in unanticipated situations requiring novel, innovative approaches. The history of space exploration is rife with examples of missions (or at least safe return of the crew) being accomplished in spite of numerous malfunctions. Indeed, most Shuttle training is preparing for the unexpected. Fewer crew-members could mean lack of either the necessary labor or the detailed expertise onboard to respond effectively to certain classes of malfunctions. Furthermore, expense and development time are required to increase equipment reliability (i.e., increased mean-time-between-failures) and minimize malfunctions.

Technology Readiness. Fewer crew members will mean increased reliance on technologies that do not currently exist. Future program managers, to meet budget and schedule, will find themselves in the unenviable position of trusting in technology development they largely have no control over.

Based on historical precedence, technology development can be pushed, but only within very finite limits. Gambling an entire program on state-of-the-art forecasts can be very risky.

Case Study 1 Modification: Two Crewmembers to Phobos. To better understand the impacts of minimizing flight crew, the crew size of an existing baseline case study was varied. Case Study 1, Human Expedition to Phobos, calls for four crew-members on a 14-month round trip to the martian moon Phobos. The question arose: "Is it possible to do the same mission with only two crewmembers?"

Table 4.15-I below shows the mass impact of four and two crewmembers for the aerobraked version of Case Study 1. Two-crewmember masses were determined by the ratios of the original masses provided by MSFC, with scaled-down crew modules, half of the crew consumables, and the same scientific payloads. (The original ratios remain valid if the delta V is the same for both crew sizes.) Essentially, the mass in LEO is decreased 25 percent. The actual cost impact is more complex, and would be a factor of the issues below.

The following issues apply to the degradation of the functional capabilities, human/machine productivity and the increased risk that may be associated with a two-person expedition to Phobos.

There is a loss of flexibility for two-shift operations during transit and while in Mars orbit. The spacecraft must be fully automated during sleep periods at all mission phases.

If one crew-member becomes incapacitated, there is an increased risk of mission abort or limited accomplishment of mission objectives since the one remaining crew member would be less likely than three to be able to perform all required mission functions. All safety-critical operations and maintenance must be designed to be automated or operable by one crewmember.

Flexibility is lost for parallel Phobos exploration and rover teleoperation while in Mars orbit, assuming a piloted two-person Phobos excursion vehicle that is separate from the orbiting, piloted mother ship. Serial time available for teleoperation of the Mars rovers would be reduced by the duration of the Phobos excursion. The mother ship must be capable of extended unpiloted operations. In this case, the excursion vehicle must also be capable of operating in an unpiloted mode if two crew members are required to perform scheduled or unscheduled Phobos surface operations.

If a single ship is used for both Mars orbit and Phobos operations, it still must be capable of automated operations when two crew-members are required to perform Phobos surface operations.

While the crew is exploring Phobos, the Mars rovers will be on their own. Therefore, they must be capable of performing at about the same level of autonomy as in the current Mars Rover Sample Return (MRSR) concept. In this case, some of the leverage associated with real-time human decision-making capability is lost. This may justify flying the MRSR separately.

Two crewmembers will be expected to spend a significantly larger portion of available time in routine and unscheduled maintenance. This function will probably outweigh science functions, limiting crewmember selection to primarily engineering experts. The lack of science expertise will significantly reduce the science return both at Phobos and for the Mars rover part of the mission.

The psychological impact of confining two people in close quarters for 14 months may be significant.

There may be a significant advantage in operational flexibility, safety, companionship, and science return by minimizing pressurized volume per crewmember, rather than lowering the number of crew.

TABLE 4.15-I.- TRADE STUDY ON 4 VERSUS 2 CREWMEMBERS FOR CASE STUDY 1: HUMAN EXPEDITION TO PHOBOS

	Masses (t)			
CARGO MISSION	Alt.A[1] (4 crew)		(2 crew)	
MCV-IMLEO[2](Mars cargo vehicle)	311.7		235.7	
Prop system / stages		27.1		20.5
Propellant		175.4		132.6
MTV		109.2		82.6
MTV (Mars transfer vehicle)	109.2		82.6	
Prop system / stages		0.0		0.0
Propellant		5.4		4.1
MOV		103.8		78.5
MOV (Mars orbit vehicle)	103.8		78.5	
Prop system / stages		3.0		3.0
Propellant		11.5		11.5
TEIS, PhEV, etc.		80.8		55.5
MOCS aerobrake		8.5		8.5
HUMAN MISSION				
MSS - Initial mass in LEO	452.9		344.7	
Prop system / stages		42.4		32.3
Propellant		280.3		213.3
MTV		130.2		99.1
MTV (Mars transfer vehicle)	130.2		99.1	
Prop system / stages		0.0		0.0
Propellant		12.7		9.7
MOV/F1 (+consum.)		117.5		89.4
MOV/F1[3] (Mars orbit vehicle)	111.1		86.2	
Prop system / stages		6.3		6.3
Propellant		36.6		36.6
MOV/Ff (+consum.-TEIS)		59.7		34.8
MOCS aerobrake		8.5		8.5
MOV/Ff[4]	119.9		70.0	
Prop system / stages		8.9		5.2
Propellant		51.9		30.3
ETV		59.1		34.5
ETV (Earth transfer vehicle)	59.1		34.5	
Prop system/ stages		1.3		1.2
Propellant		4.0		2.9
ECCV+IMM+crew eqpt/consumables		53.4		31.0
Earth crew capture vehicle	6.9		4.1	
Interplanetary mission module	43.3		25.0	

[1] Alt.A: 4 crew, aerobraking at Mars for both the human and crew flts

[2] Mars cargo vehicle-IMLEO - Initial mass in low Earth orbit of MCV

[3] Mars orbit vehicle/F1 - MOV just after entering Mars orbit

[4] Mars orbit vehicle/Ff - MOV just before leaving Mars orbit

Conclusion. Unfortunately, conclusive assessment criteria have not yet been developed for future human exploration missions. Factors to be considered include safety, short-term and long-term investment, total program costs, peak costs, development costs, and short-term and long-term benefits.

Reducing the Phobos expedition to two crewmembers results in potentially significant degradation of mission objectives (including science) and flexibility, at significantly higher risk, with a 25 percent reduction in overall mass at LEO. The associated costs would most likely not be lower than the 25 percent associated with the mass decrease, and might actually be higher due to increased reliance on new technology and higher reliability systems.

SECTION 5

Indepth Assessments

The Office of Exploration (OEXP) is supported by technologists in various disciplines assigned to provide independent assessments (from the technologists' perspective) of the case study system requirements and implementation. This support is for the purpose of identifying high-leverage technologies which might significantly enhance the efficiency and/or effectiveness of one or more case study implementation concepts. For example, by reducing mass to LEO, efficiency is increased, or by increasing component reliability or by reducing operational complexity, effectiveness improves. The following sections highlight the FY 1988 activity of the indepth systems level assessments. Table 5-I summarizes study objectives and results.

TABLE 5-I.- INDEPTH SYSTEM ASSESSMENT SUMMARY

Study/report	Affected Case Study	Objective	Results
5.1 Power Systems			
5.1.1 SP-100 Nuclear Power System Conceptual Designfor Lunar Base Applications	3	Provide a conceptual design of a nuclear power system utilizing an SP-100 reactor and a Stirling engine conversion for use on the lunar surface	(1) Nuclear power system offers substantial mass savings over comparable PV /storage power systems and enables continuous day and night operation without requiring energy storage (2) The mass of a 50 kWe solar PV power system with RFC storage for full night power exceed the mass of the entire 825 kWe nuclear power plant.
5.1.2 Solar Photovoltaic Versus Nuclear Power for Lunar Observatory	3	Compare solar photovoltaic (PV) and nuclear power systems for the construction and operations phases of a farside lunar observatory	(1) For operation power levels up to about 60 kWe, a solar photovoltaic power system was found to be attractive from construction time and system mass viewpoint (2) For operational power levels in excess of about 60 kWe, the nuclear power system exhibits a mass advantage over solar PV power systems. (PFC) storage is mass advantageous for construction times up to 42 days. If additional storage time is required, (RFC) storage would be favored.

TABLE 5-I.- Continued

Study/report	Affected Case Study	Objective	Results
5.2 Propulsion Systems			
5.2.1 Evaluation of Advanced Propulsion/Power Concepts	1, 2, 4	Evaluate the capabilities and performance potential of various nonchemical propulsion concepts and assess the leverage which such technologies could provide to future NASA initiatives	(1) Electric propulsion systems appear to be well-suited to lunar and interplanetary cargo missions where quick trip times are not a high priority; solar and laser thermal rocket concepts offer some advantages in trip time over electric propulsion at the expense of reduced payload. (2) Solid- and gas-core nuclear thermal rockets offer some of the best prospects for sprint missions for the next two decades. Beyond the 2020 yr timeframe introduction of high thrust/high Isp magnetic and inertial fusion rockets could make solar system class spacecraft a reality.
5.2.2 Impact of Solid-Core NTR Propulsion on Human Expeditions to Phobos/Mars	1, 2	Quantify the savings in initial mass in Earth orbit (IMEO) for the split cargo and piloted sprint missions using solid-core/NTR technology	Case Study 1 (1) Compared to chemical propulsion, the use of NTR technology can decrease a total IMEO by 40 - 50% over the Isp range of 850 - 950/s. (2) Increasing NTR Isp from 850 s to 950 s results in a savings of 130 t (3) Increasing engine thrust from 100 KLB to 250 KLB on the TMI stage increases the total mass by 14 t which may be offset by the improvement in thrust-to-

TABLE 5-I.- Continued

Study/report	Affected Case Study	Objective	Results
5.2.2 (Continued)			Case Study 2 At an Isp of 900 s, an all-propulsive NTR provides a total mass savings on the order of 5% ~(12% at 950 s) over the aerobraked chemical propulsion
5.2.3 Impact of Phased Implementation of Solid and Gas-Core Nuclear Thermal Rocket Propulsion On Evolutionary Lunar to Mars Outpost Case Study	4	Estimate the savings in mass in Earth orbit and the logistical simplification resulting from the introduction of increasingly more efficient gas-core/nuclear thermal rocket technology into the evolutionary case study Concepts examined include the closed cycle NLB system and the space-radiation-cooled, open cycle GCR/SRGCR concept.	Case Study 1 (1) Using NLB technology, the reference cargo/sprint missions can be performed all-propulsively from LEO with ~27% less mass than that required by the NEP cargo and aerobraked chemical systems/launch from lunar orbit (2) For approximately 1000 t in LEO, the SRGCR can perform all-up, all-propulsive class missions to Mars and back in 280 days. (3) Gas-core/nuclear thermal rocket could be available around the 2010 timeframe.
5.2.4 Issues of Mars Orbital Refueling	1,2	Identify and clarify the options and issues associated with refueling a piloted vehicle	(1) Refueling in Mars orbit preferred over all-up two-way chemical sprint vehicle (2) High Mars orbit with Deimos surface as a contingency location is preferred storage/transfer location (3) Preferred transfer options is either fluid transfer or redundant vehicles with fluid transfer capability (4) Accelerate cryogenic fluid transfer technology development for the expedition missions and develop technology for remote system status monitoring

TABLE 5-I.- Concluded

Study/report	Affected Case Study	Objective	Results
5.2.5 Launch of Cryogenic Tanks	1, 2, 3, 4	Determine the better method of launching cryogenic storage tanks and contents to LEO, either loaded storage tanks or by using separate transfer tanks	(1) Launching propellant to LEO in flight tankage could result in mass penalty between 50% for state-of-art technology to 15% with technology advances over fluid transfer option. (2) Moderate technology advance will be required for manned Mars mission that use LOX/LH2. A lunar mission would not have such a severe penalty.
5.3 Advanced Life Support Systems			
5.3.1 Life Support Architecture and Technology Requirements Definition	1, 2, 3, 4	Conceptualize a life support system (LSS) requirements and approach, including technology options to accommodate OEXP case studies	(1) Although Space Station Freedom and Shuttle technology are theoretically feasible, a significant penalty primarily associated with food and water is incurred with expendables.
5.3.2 Extravehicular Activity Requirements Definition	1, 2, 3, 4	Identify technology requirements for the EVA system necessary to complement future lunar and Mars missions	(1) For lunar EVA system, top level requirement issues have been identified for the pressure envelope, life support system, support vehicles, EVA support equipment, and airlock.

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5.1 POWER SYSTEMS

5.1.1 SP-100 Nuclear Power System Conceptual Design for Lunar Base Applications

The objective of this study was to provide a conceptual design of a nuclear power system using an SP-100 reactor and Stirling engine conversion for use on the lunar surface. System configurations were selected for their ability to enable and/or enhance a lunar base mission. Numerous system components and coupling options were examined and recommended options were chosen for safety implications, high performance, low mass, and ease of assembly.

Background. This conceptual design study was performed as a result of a request from the Propulsion, Power, and Energy Division in the Office of Aeronautics and Space Technology (OAST). The design includes system performance and sizing data, as well as layout rationale. An artist's rendering of the nuclear power system as it applies to a typical mature lunar base was included as part of the study (figure 5.1.1-1). Because of obvious implications to the Office of Exploration (OEXP) case studies, the conceptual design study was extended to provide an evaluation of nuclear power system impacts on an advanced lunar base.

Key Assumptions.

a. Mature lunar base with power requirements in the 700-900 kWe range.

b. Presence of rovers for construction and maintenance.

c. Advanced technologies, including the SP-100 reactor, free-piston Stirling engines, and mercury heat-pipe radiators.

d. Nuclear power system supplies electrical power only; the use of thermal energy from the power system will be examined in future studies.

e. Use of lunar-soil shielding designed to meet human safety requirements.

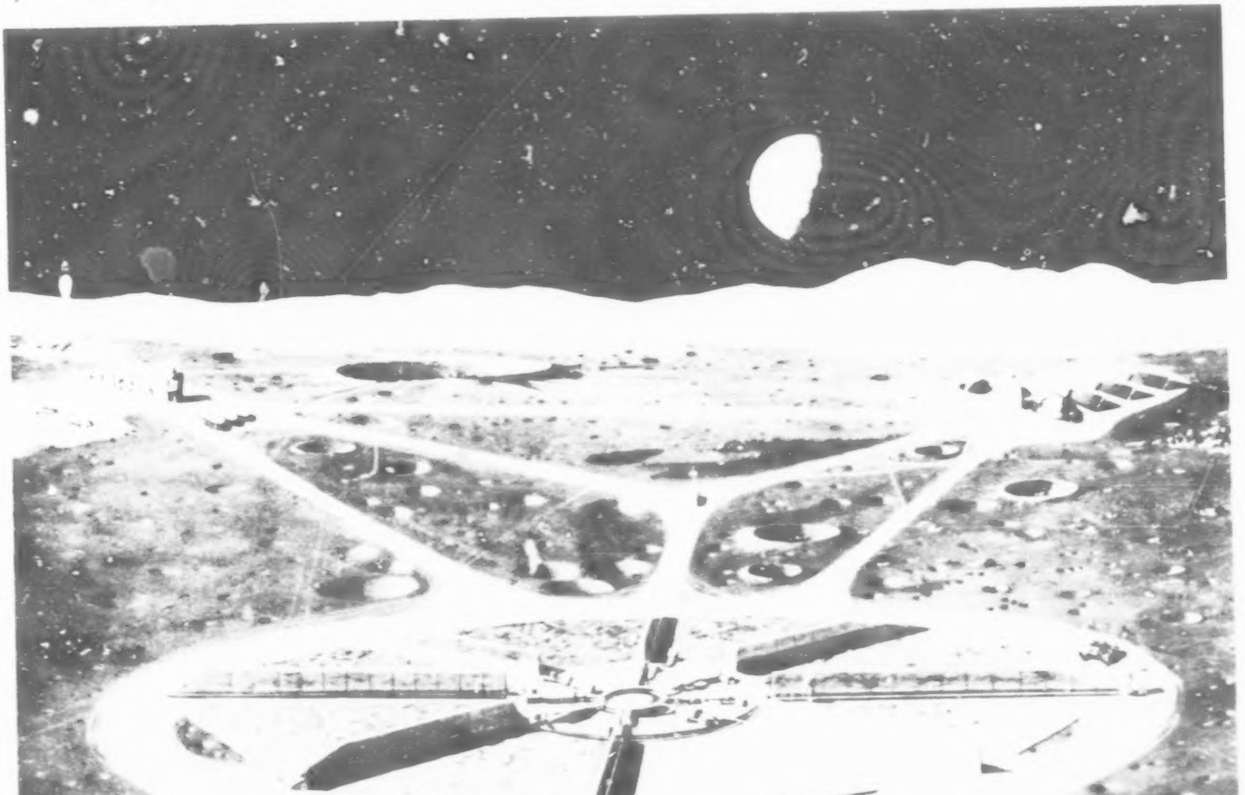

Figure 5.1.1-1. Artist's rendering of a nuclear power system for lunar base applications.

Approach. This conceptual design was developed using Lewis Research Center (LeRC) experience with the SP-100 reactor and Stirling engines. A critical aspect of the study was to gain an understanding of the lunar environmental conditions and to identify their impacts on the design of the power system. To more fully understand the interactions of a reactor power system with a lunar base, a possible mature lunar base concept was defined with the assistance of the OEXP Surface Systems Integration Agent. The study was a 3-month in-house effort of LeRC's Advanced Space Analysis Office (ASAO) and Power Technology Division (PTD).

Findings.

Lunar Base Assumptions. The lunar base concept is derived from studies performed at the Johnson Space Center. The central core of the base is comprised of two inflatable, spherical modules for habitation and scientific research. These modules are partially buried and shielded from cosmic radiation. Adjacent to the inflatable structures is a rover storage and recharging facility. The inflatable modules and rover facility can be seen in figure 5.1.1-1 in the upper right corner.

A lunar soil processing plant producing oxygen is located approximately 5 kilometers from this habitation area. Lunar soil is transported to this plant after it has been collected from a nearby mining site. A launch and landing facility is located within a kilometer of the oxygen plant. The proximity of the launch pad to the processing plant enables oxygen for propellant to be delivered quickly to orbit for use in chemically propelled transfer vehicles. The processing plant and landing pad are located in the upper left corner of the figure.

Power Requirements. A solar photovoltaic (PV) power system with regenerative fuel cells is assumed to meet the power requirements of the initial habitat module (25 to 100 kWe). As the base expands to include scientific experimentation, rover recharging, and soil processing, a nuclear power system becomes the most viable means of meeting the higher power requirements.

The processing plant will be the predominant power load. For the purpose of this study, that requirement takes the form of electrical power for electrolysis of water, following a hydrogen reduction of lunar ilmenite. The thermal energy requirements of the processing plant would also be provided by the nuclear power system through electrical resistance heating. For an oxygen production capability of 25 t/m, the plant would require 740 kWe, or 90 percent of the nuclear system output.

The remainder of the electrical power generated by the nuclear system would be distributed to the habitat and science modules for life support, to the science laboratory for experimentation, and to the rover storage and recharging facility.

Nuclear Power System Design. The nuclear power system is designed with an emphasis on safety and reliability. It is shown in the foreground of figure 5.1.1-1. This conceptual design consists of a 2500 kWe SP-100 reactor coupled to eight free-piston Stirling engines. The reactor is identical to the design currently baselined in the SP-100 program, whereas the Stirling engines replace the thermoelectric power conversion system of the present SP-100 design. Two of the Stirling engines are held in reserve to provide engine backup for dependable power generation. The remaining six engines operate at 91.7 percent of their rated capacity of 150 kWe. The design power level for this system is 825 kWe. The system is modular and can be replicated in increments of 825 kWe to meet higher power requirements.

It would also be possible, and perhaps desirable, to replicate this system design and operate the two systems at reduced power levels to meet the 825 kWe power requirement. If one reactor power system needs to be shut down, the other system could compensate for the loss in power. As power requirements increase, the capacity of the systems could be gradually increased to meet the higher power levels.

The Stirling engines are arranged in a spoked-wheel configuration and share a common heat transport manifold with the reactor. Each engine is equipped with a pumped heat-rejection loop connected to a mercury heat-pipe radiator. The radiator panels are arranged in a vertical configuration and extend radially from the Stirling engines. A thermal apron is placed between the panels to reduce the lunar surface temperature and thus reduce the required radiator area. The total mass of the system, including power conditioning and transmission lines, is 20 t.

The reactor is located in an excavated cylindrical hole which provides shielding from gamma and neutron radiation. The use of lunar soil eliminates the need to transport heavy terrestrial shielding materials to the lunar surface. A boral bulkhead with a domed cap maintains a dust-free environment for the reactor.

Safe radiation levels are maintained in all directions around the power system. This allows for flexibility in choosing a reactor site. The excavated shield design also allows for periodic maintenance on the system's radiator panels. For this conceptual design, the nuclear power system has been placed 1 km from the habitation area and approximately 4 km from the processing plant.

OEXP Impacts. Nuclear power has many impacts on the operations of a lunar base. It provides a substantial amount of power for a variety of lunar [illegible] such as large scale processing of lunar materials, expanded scientific experimentation, and/or the use of a closed-loop life support system. This nuclear power system conceptual design offers a substantial mass savings over comparable PV/storage power systems. For example, the mass of a 50-kWe solar PV power system with regenerative fuel cell storage for full night power capability exceeds the mass of the entire 825-kWe nuclear power plant. The nuclear system also enables continuous day and night operations without the need for energy storage.

Potential FY 1989 Activities. General studies that would contribute to future extraterrestrial nuclear technology are

a. A conceptual design for nuclear power system supply of thermal energy, either from the reactor or from waste heat rejection, for lunar material processing

b. A chronological description of lunar base power system evolution based on power requirements and technology readiness

c. A conceptual design of a similar nuclear power plant on the surface of Mars

Component Studies/Tradeoffs. Potential studies and trades related to specific systems components might include

a. The tradeoffs involved in replacing the mercury heat pipes with lower temperature or high pressure water heat pipes for waste heat rejection

b. A study of the power conditioning and distribution components associated with this conceptual design

c. Transmission line mass as a function of power system location for various base layouts and distribution forms

d. The safety and technological implications associated with transmission line deployment for buried and suspended lines

5.1.2 Solar Photovoltaic Versus Nuclear Power For Lunar Observatory

The objective of this assessment was to compare solar photovoltaic (PV) and nuclear power systems for the construction and operations phases of a far-side lunar observatory outpost and to document significant results and findings.

Background. The baseline of the OEXP case study was to emplace and operate a moderately sophisticated complement of scientific observational instrumentation on the far side of the moon. The baselined ground rules for this case study are that the setup of the observatory will be accomplished over a 2-year period, beginning in the year 2004, with one cargo and one crew mission per year. Crew staytimes for construction are baselined at 14 days per trip or less. The lunar observatory will also be operating unattended for long periods of time; therefore, the power system selected must show high reliability and autonomy. A comparison of total power system masses between solar PV and nuclear power systems was made for the observatory construction and operations phases.

Key Assumptions. The construction-phase power requirement is assumed at 20 to 40 kWe; the operations-phase power requirement is assumed at 50 to 100 kWe. These requirements were obtained from the OEXP Surface Systems Integration Agent.

Amorphous silicon solar PV arrays that could be easily rolled out on the lunar surface in a few days were assumed to supply construction power requirements. If the construction phase were extended into the lunar night, energy storage in the form of primary hydrogen-oxygen fuel cells (PFC's) or regenerative hydrogen-oxygen fuel cells (RFC's) was assumed in addition to the PV arrays.

The nuclear power system [illegible] assumed at [illegible] heat [illegible] shielding [illegible]. Construction time for the nuclear power system was assumed to be greater than 14 days and thus precluded the use of such a system as a power source for the observatory construction phase. [illegible] phase of the observatory [illegible] system would be supported by solar arrays [illegible] energy storage [illegible] be required [illegible]. The mass [illegible] arrays and the energy storage [illegible] with the mass of the [illegible] power system [illegible] phase.

Approach. [illegible]

Findings. [illegible]

system-mass viewpoint. This solar PV power system consists of amorphous silicon rollout arrays to provide initial power. Sun-tracking gallium arsenide foldout arrays and RFC storage stacks, gaseous reactant tanks, radiators, and power management and distribution equipment are then deployed/erected. The final stage of construction would consist of connecting the initial rollout amorphous silicon arrays to the RFC's for nighttime operation. The sun-tracking foldout gallium arsenide PV arrays provide constant daytime operational power to the lunar observatory. The complete operational power system should be erectable within one 14-day staytime. However, once the RFC's are operational, extension of the power system construction period into the lunar night is possible if additional construction time is required.

For operational power levels in excess of about 60 kWe, the nuclear reactor power system exhibits a mass advantage over the solar PV power system, and that mass advantage increases significantly with higher power requirements. This concept includes longer construction times than those required for the solar PV power system. As stated previously, this solar PV power system would use amorphous silicon rollout PV arrays for initial power and either a primary or regenerative fuel cell storage system. PFC storage is mass-advantageous for construction times of up to 42 days. but if additional storage time is required, RFC storage would be more favorable.

Issues/Open Items. A second nuclear reactor power plant option was identified, but the scope of the study precluded more than a cursory treatment at this time. The concept does not require a solar PV power system for construction and can provide lunar observatory operational power well within a 14-day staytime. It is based on locating a completely assembled and fully shielded SP-100 reactor thermoelectric power system within a lunar lander. Radiators are deployed from the lander at the surface site, a power cable is installed, and the power system is fully operational within 24 hours of cable connection to the load. Depending on the lander power capability, an amorphous silicon PV array could be initially deployed on the lunar surface to provide power if required. This concept has not been studied in as much detail as the SP-100 surface concept which was evaluated for high power lunar base applications. However, it shows about a 20 percent mass advantage over a surface nuclear power unit at the 40 kWe power level. A potential issue for the nuclear lander option is the volume and mass constraint imposed by the lunar lander. However, this concept is very similar to the existing SP-100 100 kWe design for orbital applications which can be stowed into a volume equivalent to one-third of the Shuttle cargo bay.

Planned or Required FY 1989 Activity. In FY 1989, a reevaluation of construction timelines for the Lunar Observatory Case Study should be made for erection of both scientific instruments and power systems. A determination should be made as to whether the lunar observatory could evolve into a larger manned base with an increased power level requirement.

It is recommended that an evaluation be made of the number of missions that would be required and of crew staytime per mission for the construction phase for both PV and nuclear power systems. Ancillary to this, a study should be performed trading the total mass to the lunar surface as well as the launch mass to LEO for each power system.

Depending on the location and dispersion of the scientific packages on the lunar surface, dispersed, smaller power systems at various science sites may be better than a centralized larger power system with long transmission line requirements. It is recommended that a trade study investigating dispersed- versus-centralized power systems be performed for this application.

Finally, rapid deployment of an SP-100 system employing thermoelectric conversion would allow more time for the construction of scientific instruments. Therefore, a scoping study should be made to address the feasibility of installing and operating an SP-100 reactor thermoelectric system in a lunar lander.

5.2 PROPULSION SYSTEMS

5.2.1 Advanced Propulsion/Power Concepts

The objective of this short study was to evaluate the capabilities and performance potential of various nonchemical propulsion concepts and to assess the leverage (e.g., lower initial mass in Earth orbit (IMEO), quicker trip times, higher payloads) which such technologies could provide to future NASA initiatives.

Approach. A successive screening/filtering process (figure 5.2.1-1) was used to identify and compare "credible" propulsion concepts on the basis of:

a. Technology maturity

b. Performance potential

c. Mission compatibility issues

This screening process was dictated by the existence of a large number of propulsion concepts (table 5.2.1-I). These concepts can use primary power either directly to generate thrust (as in the case of chemical rockets or solar sails) or indirectly by converting power to electricity for use with electric propulsion (EP). Similarly, a large number of existing power sources and conversion techniques require examination.

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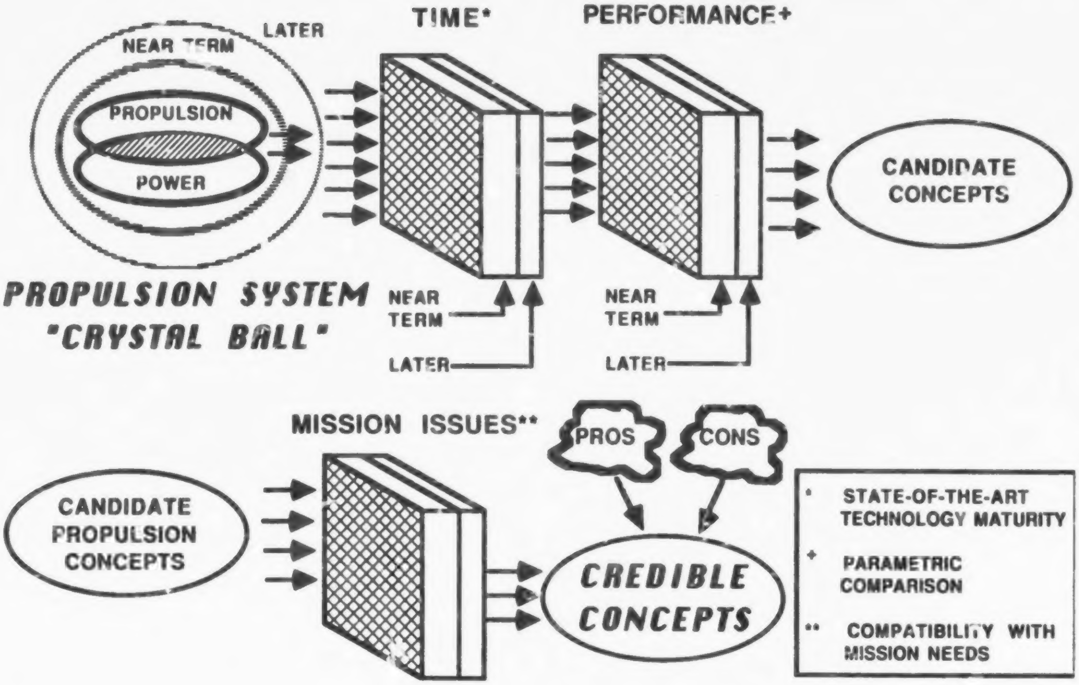
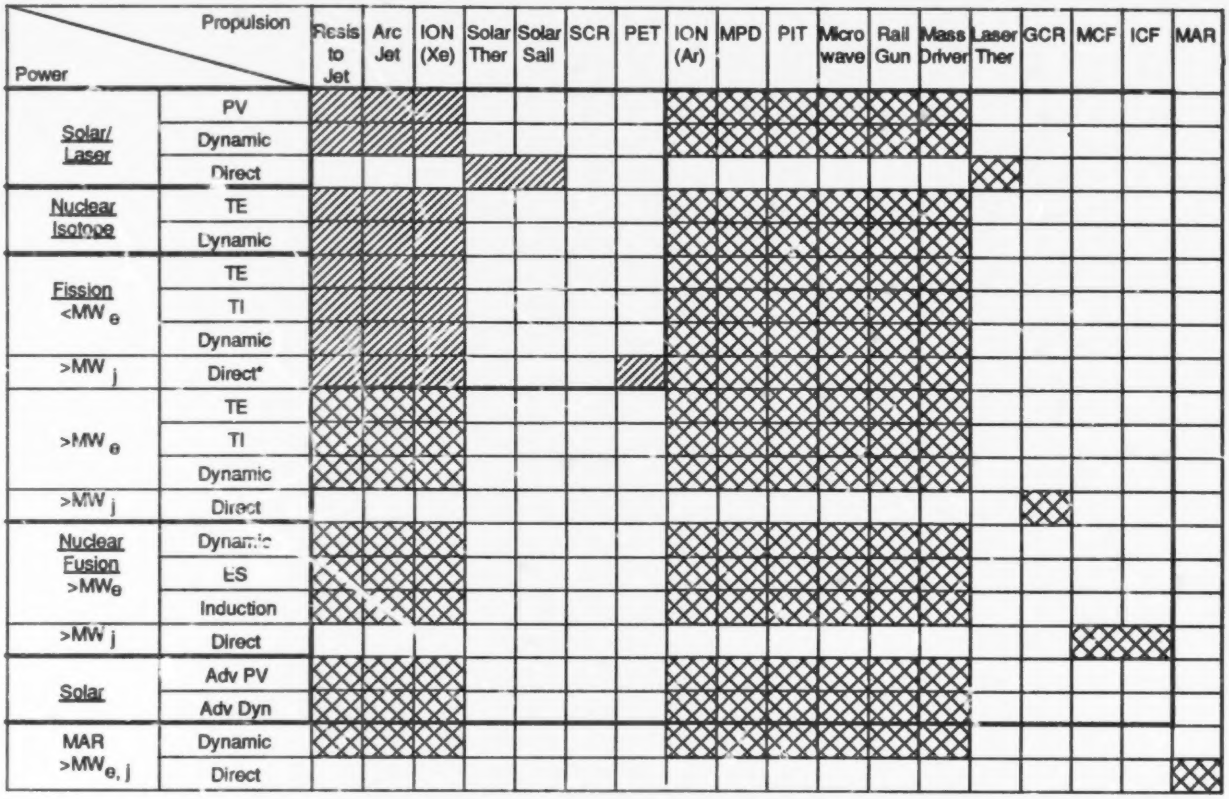

NEAR TERM
LATER
PROPULSION
POWER
TIME*
PERFORMANCE+
CANDIDATE CONCEPTS
PROPULSION SYSTEM "CRYSTAL BALL"
NEAR TERM
LATER
NEAR TERM
LATER
MISSION ISSUES**
PROS
CONS
CANDIDATE PROPULSION CONCEPTS
CREDIBLE CONCEPTS
* STATE-OF-THE-ART TECHNOLOGY MATURITY
+ PARAMETRIC COMPARISON
** COMPATIBILITY WITH MISSION NEEDS

Figure 5.2.1-1.- Filtering approach used in study.

TABLE 5.2.1-I.- POWER AND PROPULSION CONCEPTS CONSIDERED IN EVALUATION

Power		Propulsion
Solar		Electric Propulsion (EP)
	Photovoltaic (PV)	Resistojet, Arcjet, Microwave Thruster
	Dynamic (SD)	Ion
		Pulsed Electothermal Thruster (PET)
Nuclear		Magnetoplasmadynamic Thruster (MPD)
Isotope	Thermoelectric (TE)	Pulsed Inductive Thruster (PIT)
	Dynamic	Rail Gun
		Mass Driver
Fission	Thermoelectric (TE)	
	Thermionic (TI)	Solar Sail
	Dynamic	Solar Thermal Rocket (STR)
		Laser Thermal Rocket (LTR)
Fusion	Dynamic	
	Electrostatic (ES)	Nuclear Fission
	Induction	Solid Core Rocket (SCR)
		Gas Core Rocket (GCR)
Mass Annihilation	Dynamic	
		Nuclear Fusion
		Magnetic Confinement Fusion (MCF)
		-Tokamak Fusion Rocket (TFR)
		Inertial Confinement Fusion (ICF)
		- Inertial Fusion Rocket (IFR)
		- Livermore IFR concept (Vista)
		Mass Annihilation Rocket (MAR)



Power	Propulsion	Resisto Jet	Arc Jet	ION (Xe)	Solar Ther	Solar Sail	SCR	PET	ION (Ar)	MPD	PIT	Micro wave	Rail Gun	Mass Driver	Laser Ther	GCR	MCF	ICF	MAR
Solar/ Laser	PV	▨	▨	▨					☒	☒	☒	☒	☒	☒					
	Dynamic	▨	▨	▨					☒	☒	☒	☒	☒	☒					
	Direct				▨	▨									☒				
Nuclear Isotope	TE	▨	▨	▨					☒	☒	☒	☒	☒	☒					
	Dynamic	▨	▨	▨					☒	☒	☒	☒	☒	☒					
Fission <MW $_e$	TE	▨	▨	▨					☒	☒	☒	☒	☒	☒					
	TI	▨	▨	▨					☒	☒	☒	☒	☒	☒					
	Dynamic	▨	▨	▨					☒	☒	☒	☒	☒	☒					
>MW $_j$	Direct*	▨	▨	▨				▨	☒	☒	☒	☒	☒	☒					
>MW $_e$	TE	☒	☒	☒					☒	☒	☒	☒	☒	☒					
	TI	☒	☒	☒					☒	☒	☒	☒	☒	☒					
	Dynamic	☒	☒	☒					☒	☒	☒	☒	☒	☒					
>MW $_j$	Direct															☒			
Nuclear Fusion >MW$_e$	Dynamic	☒	☒	☒					☒	☒	☒	☒	☒	☒					
	ES	☒	☒	☒					☒	☒	☒	☒	☒	☒					
	Induction	☒	☒	☒					☒	☒	☒	☒	☒	☒					
>MW j	Direct																☒	☒	
Solar	Adv PV	☒	☒	☒					☒	☒	☒	☒	☒	☒					
	Adv Dyn	☒	☒	☒					☒	☒	☒	☒	☒	☒					
MAR >MW$_{e, j}$	Dynamic	☒	☒	☒					☒	☒	☒	☒	☒	☒					
	Direct																		☒

*Note: Direct nuclear rockets can be integrated with power conversion and EP to obtain a "hybrid" system with the advantages of both systems.

▨ Near term (10-15 years) ☒ Later term (>15 years)

Figure 5.2.1-2.- Propulsion concepts filtered according to time.

Based on the maturity of the technology and its anticipated availability date, concepts were categorized as either a near-term or longer-term technology. A matrix chart listing the various propulsion/power concepts and reflecting this time filter is shown in figure 5.2.1-2. The relative performance potential of these concepts was also examined using key performance parameters that included specific impulse, propulsion system specific mass, and engine thrust-to-weight. Systems with specific impulses less than or equal to state-of-the-art chemical propulsion (Isp = 500 s) were eliminated for primary propulsion applications. For propulsion systems with comparable specific impulse capability, those concepts with significantly higher specific masses were also eliminated from consideration.

Mission/propulsion system compatibility issues provided a final screen for identifying attractive candidate concepts. NASA has identified the establishment of a permanent lunar base and piloted missions to Mars as potential future initiatives. To support these initiatives, propulsion concepts with high specific impulse and high spacecraft thrust-to-weight are highly desirable. For the most part, near-term technologies such as chemical and electric propulsion have one but not both of the desired attributes mentioned above. In the future, "high leverage" technologies may be available which will allow large quantities of cargo to be transported quickly over interplanetary distances. A sampling of propulsion system designs ranging from near-term nuclear electric propulsion (NEP) systems to solar-system-class inertial fusion rockets are listed in table 5.2.1-II and their compatibility with sprint and cargo class missions are shown in figure 5.2.1-3.

Findings. In general, the EP systems occupy a region of parameter space where the specific impulse and mass are ~2 to 10 kiloseconds and ~10 to 50 kg/kWj, respectively. With an engine thrust-to-weight of ~10^{-4}, EP systems appear to be well-suited to lunar and interplanetary cargo missions when quick trip times are not a high priority. Solar and laser thermal rocket concepts offer some advantages in orbital transfer vehicle (OTV) trip time over EP systems, but at the expense of reduced payload fraction.

TABLE 5.2.1-II.- SAMPLING OF PREVIOUSLY STUDIED ENGINE DESIGNS

	System	Destination	Trip time*	Payload mass† / Initial mass
1)	NEP/ION (SP-100)	Neptune	12 Years (Probe-1 year)	0.10
2)	LTR (1 MW_t)	LEO-GEO	28 Days	0.44
3)	STR (2 MW_t)	LEO-GEO	30 Days	0.54
4)	NEP/ION (300 KW_e)	Moon	370 Days	0.57
5)	Pegasus/MPD (8.5 MW_e)	Mars	1000 Days	0.45
6)	NEP/ION (300 KW_e)	NSO Mars	770 Days (Cargo-1 way)	0.44
7)	NEP/ION (3 MW_e)	Moon-Mars	413 Days (Cargo-1 way)	0.54
8)	SCR/NERVA ((5000 MW_t)	Mars	720 Days	0.23
9)	NEP/ION (400 MW_e)	Mars	- 180 Days	0.0
10)	GCR (8500 MW_t)	Mars	90 Days / 280 Days	0.075 / 0.25
11)	Orion (43,000 MW_t)	Mars	250 Days	0.23
12)	TFR (7500 MW_t)	Mars	77 Days	0.06
13)	Vista (225,000 MW_t)	Mars	100 Days	0.017
14)	IFR (200,000 MW_t)	Mars	55 Days / 20 Months	0.26 / 0.14

* Round trip unless otherwise indicated

† To destination only

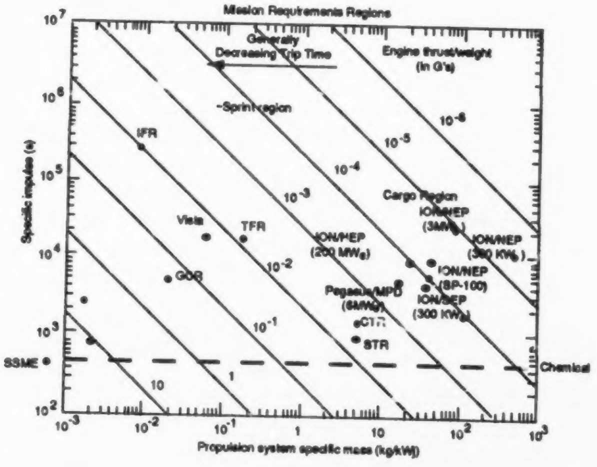


Figure 5.2.1-3.- Performance regions of mission / propulsion system compatibility.

It is difficult for EP systems to operate in the sprint region. A 200-MWe ion/NEP system, with a specific mass and impulse of ~1 kg/kWj and ~20,000 s, respectively, was examined by the Jet Propulsion Laboratory (JPL) for its quick trip potential. The system was capable of a 7.5-month round-trip mission to Mars, but its initial mass was ~1500 t and the propellant and payload fractions were ~80 percent and 6 percent, respectively. At a 400-MWe power level, 6-month round-trip times could be achieved, but only for a zero payload fraction.

Of the various concepts capable of being developed over the next two decades, solid and gas core nuclear thermal rockets offer some of the best prospects for sprint missions. Solid core technology and significant research into gas core feasibility issues were demonstrated during the nuclear energy for rocket vehicle application (Nerva) program, adding to the technical maturity level of these concepts.

In summary, a large number of propulsion system options are available for cargo applications but few options exist for quick sprint missions (figure 5.2.1-4). High-power solar and laser thermal concepts and advanced technology solar/nuclear EP (~1-5 kg/kWj) may enable these concepts to break into the sprint-class region. However, solid and gas core nuclear concepts (and potential hybrid configurations) appear to be the leading contenders in this category. Beyond the 2020 timeframe, the introduction of high-thrust/high-Isp magnetic and inertial fusion rockets could make solar-system-class spacecraft a reality.

5.2.2 Impact of Solid Core Nuclear Thermal Rocket (SC/NTR) Propulsion on Human Expeditions to Phobos/Mars

The objective of this trade study was to quantify the savings in initial mass in Earth orbit (IMEO) for the split cargo and piloted-sprint missions obtained using SC/NTR technology developed and demonstrated during the Nerva program (which lasted 18 years and cost $1.5 billion).

Background. NASA's interest in NTR's dates back to 1960 when a joint Atomic Energy Commission (AEC)/NASA Space Nuclear Propulsion Office (SNPO) was established to pursue development of a Nuclear Engine for Rocket Vehicle Application (Nerva). The Nerva program had as its objective the development of a flight engine which could provide twice the specific impulse of the best chemical rockets. This technology was essentially in hand when changing national priorities forced termination of the Nerva program in January 1973. Nevertheless, the program was judged to be a technical success. Nineteen reactors were built and tested at power

Power	Propulsion	Resis to Jet	Arc Jet	ION (Xe)	Solar Ther	Solar Sail	SCR	PET	ION (Ar)	MPD	PIT	Micro wave	Rail Gun	Mass Driver	Laser Ther	GCR	MCF	ICF	MAR
Solar/ Laser	PV																		
	Dynamic																		
	Direct				▨										☒				
Nuclear Isotope	TE																		
	Dynamic																		
Nuclear Fission <MW$_e$	TE																		
	TI																		
	Dynamic																		
>MW $_j$	Direct *			▨				▨		☒	☒	☒							
>MW $_e$	TE									☒	☒	☒							
	TI									☒	☒	☒							
	Dynamic									☒	☒	☒							
>MW $_j$	Direct															☒			
Nuclear Fusion >MW$_e$	Dynamic									☒	☒	☒							
	ES									☒	☒	☒							
	Induction									☒	☒	☒							
>MW $_j$	Direct																☒	☒	
Solar	Adv PV									☒	☒	☒							
	Adv Dyn									☒	☒	☒							
MAR >MW$_{e,\,j}$	Dynamic																		
	Direct																		

* Note: Direct nuclear rockets can be integrated with power conversion and EP to obtain a "hybrid" system with the advantages of both systems.

▨ Near term (10-15 years) ☒ Later term (>15 years)

Figure 5.2.1-4.- Candidate concepts enabling quick missions in Earth orbit, cislunar, and/or interplanetary space.

levels ranging from ~1100 MWt for the nuclear reactor experiment (NRX) series to ~4100 MWt for the Phoebus-2A nuclear rocket reactor. Designed to produce a nominal thrust of 1110 kN (250,000 lbf) at an Isp of 840 s, Phoebus-2A was intended to demonstrate the feasibility of Nerva-2, a 200,000 to 250,000-lbf-class engine under consideration by NASA for use in manned missions to Mars.

The Nerva program also demonstrated sustained engine burn capability with the NRX-A6 reactor operating for over an hour at rated conditions of 1125 MWt and an equivalent Isp of 730 s. Nuclear and non-nuclear flight components were fully integrated into the experimental flight engine prototype (XE-P) system, shown in figure 5.2.2-1. Tested in the spring of 1969, the XE operated at a power level of 1140 MWt and produced a nominal thrust of 245 kN (55,000 lbf), which could be throttled from ~50 to 100 percent at full specific impulse. It was also successfully started a record 24 times and accumulated a total of 115 min of powered operation. A number of candidate control concepts under consideration for the Nerva flight engine were also evaluated on the XE, and completely automatic startup capability was demonstrated. The XE test series proved rather convincingly that a nuclear rocket engine could be started, operated, shut down, and restarted over a wide range of reactor conditions that could be encountered in flight. Advanced composite and pure carbide fuel element designs were also developed during the program for improving engine lifetime (from ~1 to 10 h at rated power) and performance (from 850 to ~950 s).

Trade Studies.

Trade No.	Description
1	Baseline - Quantify the mass savings over chemical propulsion for both the cargo and piloted sprint vehicles obtained from using SC/NTR (Isp = 900 s) propulsion.
2	Assess the impact on IMEO of varying NTR specific impulse. An Isp range of 850 to 950 s is examined.

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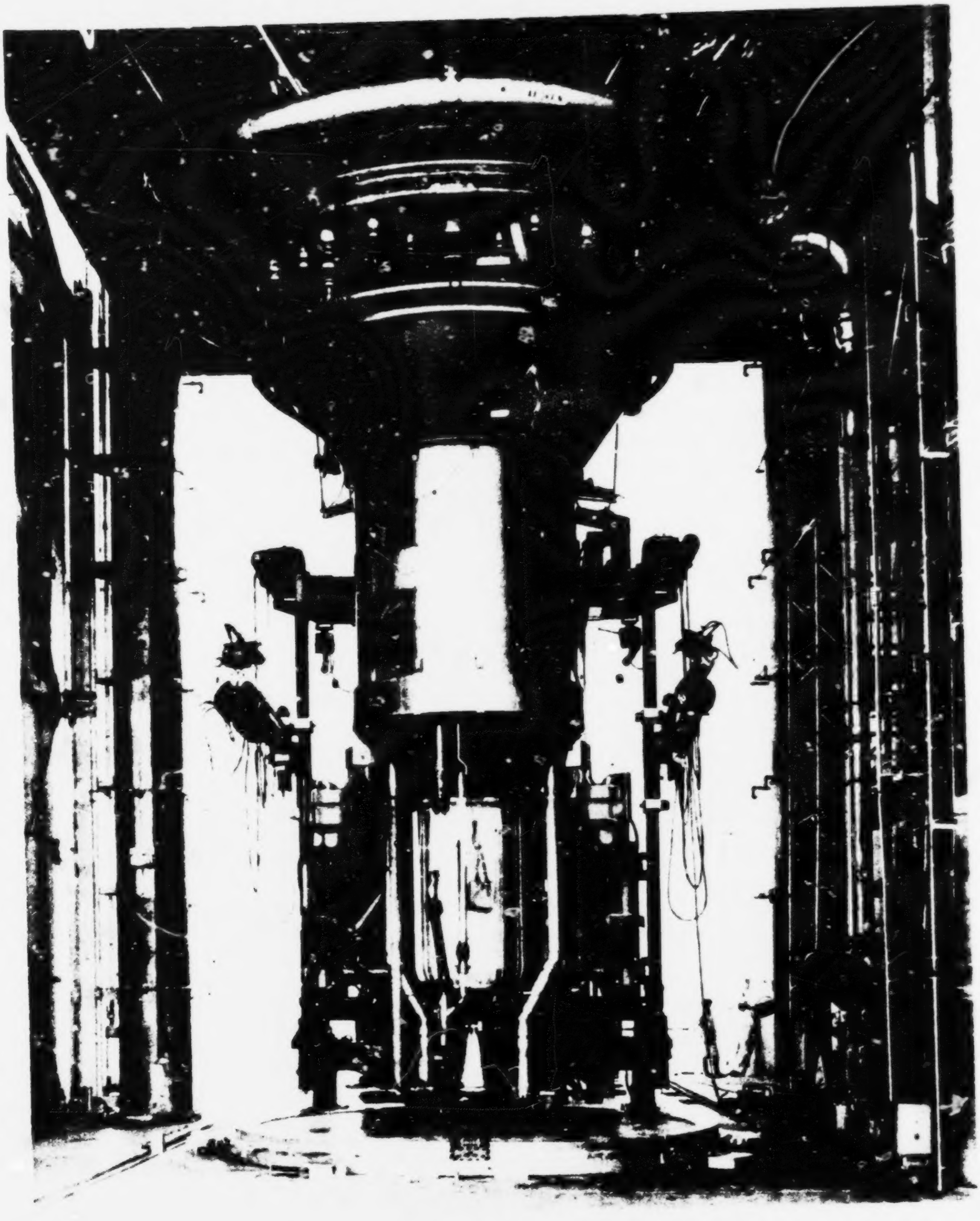

Figure 5.2.2-1.- View of the XE in the ETS-1 Facility.

3 Quantify the mass penalty associated with using higher thrust (250 vs 100 klbf) NTR engines in the Trans-Mars injection (TMI) stage for enhancing initial thrust-to-weight (F/Wi) from ~0.1 to 0.2.

4 Assess the impact on IMEO of decreasing the cooldown propellant inventory and increasing Isp for the expendable stages.

Key SC/NTR Assumptions.

a. Nerva (Isp = 850 to 900 s) and "Nerva-derivative" (Isp = 900 to 950 s) propulsion technology is available.

b. SC/NTR parameters:
   (1) F = 75 to 100 klbf, M_{eng} = 20.5 t (includes a 9-t external disk shield for added crew radiation protection)
   (2) F = 250 klbf, M_{eng} = 27.1 t (including the 9-t disk shield)

c. The remaining SC/NTR stage mass (tankage, structure, etc.) is set at 16 percent of the total propellant load.

d. Propellant penalties of 6 and 12 percent (for expendable and reusable vehicles, respectively) have been assessed against the NTR for consumption during the transient/cooldown phases. No credit is taken for the cooldown impulse provided by the NTR (a pessimistic assumption). A 5 percent propellant reserve is also used in this analysis.

e. For the Mars case study comparison we conservatively assume that only the "all-propulsive" option is available to the NTR (no aerobrakes utilized).

Analytical Approach. The "all propulsive" delta V budget, developed by SAIC for the cargo and piloted missions to Phobos (table 5.2.2-I), is also used in this study for the NTR systems. For our Mars study we have selected the 2009 piloted sprint opportunity because of its demanding delta V requirements. The NTR delta V budget for this opportunity is the same as that used by the aerobraked chemical system, with the exception that Mars orbital capture (MOC) into the reference orbit (250 km by 1 sol) is done all propulsively (ΔV_{MOC} = 3.85 km/s). For the 2007 conjunction cargo mission the delta V budget without aerocapture (w/o AC) is used.

To obtain IMEO estimates for the cargo and piloted vehicles, propellant loading must be determined. Using the specified payloads and NTR scaling assumptions, and knowing the time history of the vehicle mass, the propellant requirements are estimated using the Rocket Equation.

TABLE 5.2.2-I.- CASE STUDIES 1 AND 2 DELTA V[1] BUDGET SUMMARY

Mission	ΔV_{TMI}	ΔV_{MCC}	ΔV_{DSM}	ΔV_{MOC}[3]	ΔV_{PRM}	ΔV_{TEI}[3]	ΔV_{MCC}
2001 conjunction cargo[2]	3.735	0.050	—	1.588	—	—	—
2002 sprint w/ Venus swingby[2]	4.197	0.050	—	3.938	—	2.305	0.050
2007 conjunction cargo w/o AC	3.765	0.050	—	0.831	—	—	—
w/ AC	3.730	0.050	—	—	0.015	—	—
2009 sprint w/DSM	5.692	0.050	1.173	3.854	—	3.766	0.050

[1] ΔVs in km/s (provided by J. McAdams - SAIC)

[2] With propulsive capture at Mars arrival (Case Study 1 - Phobos mission)

Note: Mars orbital capture (into 250 km by 1 sol orbit)
All vehicles are expendable
Direct entry at Earth return assumed

[3] The ΔV_{MOC} and ΔV_{TEI} values represent coplanar capture/departure impulses into/from a 250 by 33,840 km parking orbit about Mars. These data do not account for the physical requirement to rotate the line of apsides of the parking orbit sometime between Mars arrival and departure (see section 2.1.1.2).

Findings: Phobos Mission.

a. Trade #1 - Compared to chemical propulsion, the use of NTR technology for the "all-propulsive" Phobos mission results in a 44 percent decrease in total IMEO (table 5.2.2-II). Approximately 50 percent of the 542 t mass savings is attributed to reduced propellant consumption by the piloted vehicle (from 615.1 t down to 270.7 t). The propellant requirements for the single-stage cargo vehicle are also reduced by over 50 percent compared to the chemical case. With a propellant loading of ~136 t, the cargo vehicle very closely resembles NTR stages studied in detail by NASA contractors during the 1960's and early 1970's for lunar and interplanetary applications (figure 5.2.2-2). Logistics for the Phobos mission are also simplified using NTR technology. Instead of five vehicles/stages, only three are required.

b. Trade #2 - Increasing the Isp from 850 to 950 s provides a further increase in total mass savings of ~130 t. At 950 s the IMEO is ~49 percent of the reference chemical results. Total engine burn time for the 250,000 lbf class NTR used on the cargo vehicle is on the order of 15 min. The Phoebus-2A rocket/reactor operated at a thrust level of 200,000 lbf for ~12.5 mins during its full-power test in 1968.

c. Trade #3 - Increasing the engine thrust level from 100 to 250 klbf on the TMI stage increases the total mass of the piloted vehicle by only 14 t (from 415.4 t to 429.4 t for Isp = 900 s). The spacecraft thrust-to-weight is increased, however, by more than a factor of 2, and the engine burn time for the TMI stage is reduced from 51.0 min to 21.5 min, reducing gravity losses during the TMI maneuver.

Findings: Mars Mission.

Trades 1, 2, & 4 - At Isp = 900 s, the "all-propulsive" NTR option still provides a total mass savings on the order of 5 percent (72 t) over the aerobraked chemical results (table 5.2.2-III). This savings is attributed totally to the cargo vehicle; the mass of the piloted vehicle is 105 t more

TABLE 5.2.2-II.- PHOBOS MISSION RESULTS USING NTR PROPULSION

Parameters	Isp = 850 s	Isp = 900 s	Isp = 950
2-stage piloted vehicle			
Stack mass at launch, t	470.7	429.4	396.4
Dry/propellant mass, t	103.6/306.1	97.8/270.7	93.1/242.3
F/W_i & Γ_{burn}(min)-250 klbf	0.24 & 23.2	0.26 & 21.5	0.29 & 20.0
F/W_i & Γ_{burn}(min)-100 klbf	0.10 & 56.2	0.11 & 51.9	0.12 & 48.4
Single-stage cargo vehicle			
Mass at launch, t	282.3	252.4	228.8
Dry/propellant mass, t	58.4/158.6	54.4/136.2	51.2/118.8
F/W_i & Γ_{burn}(min)-250 klbf	0.40 & 12.9	0.45 & 11.6	0.50 & 10.7
Mass savings, t			
- Cargo vehicle	146.7	176.6	200.2
- Piloted vehicle	323.9	365.2	398.2
Total mass savings, t	470.6	541.8	598.4
Percentage of ref.	38.5	44.3	48.9

Ref. SAIC Nos.: 3 - Stage piloted vehicle mass, IMEO/dry/propellant = 794.6/118.5/615.1 t
(Isp = 460 s) 2 - Stage cargo vehicle mass, IMEO/dry/propellant = 429.0/59.9/293.1 t

Note: Above referenced SAIC nos. are preliminary estimates which will be substantiated in subsequent study analysis.

Note: Total burn time for all NTR engines <1 hr. (already demonstrated)

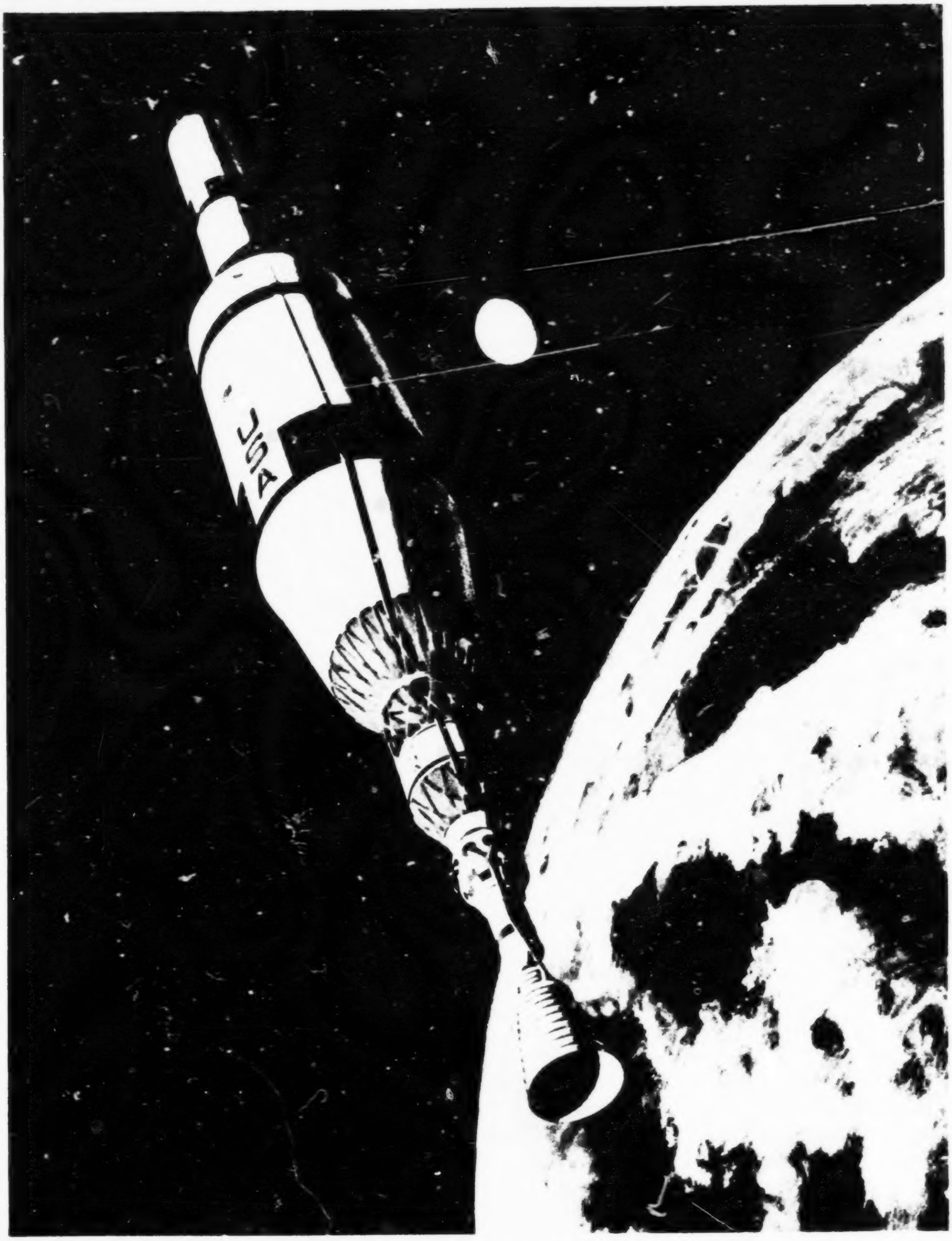

Figure 5.2.2-2.- Possible configuration for an SC/NTR staged used for propelling cargo or piloted vehicles.

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TABLE 5.2.2-III.- MARS MISSION RESULTS USING NTR PROPULSION

Parameters	Isp = 900 s		
Single stage cargo vehicle mass at launch, t	623.0		
F/W_i & Γ_{burn}(min)-250 klbf	-0.18 & 28.9		
Cargo vehicle mass savings, t	177.2		
Piloted vehicle mass at launch, t	275.7		
TMI stage sensitivity to cooldown propellant fraction & Isp		Isp = 950s	
	6% & 900 s	6% & 950 s	0% & 950 s
TMI stage mass at launch, t	584.6	534.3	465.4
Total piloted stack mass at launch, t	860.3	810.1	741.1
F/W_i* & Γ_{burn}(min)-250 klbf	0.26 & 27.0	0.28 & 25.8	0.31 & 23.6
Piloted stack mass savings, t	-105.6	-55.4	13.6
Total mass savings, t	71.6	121.8	190.8

Ref. LESC Nos.: 3-Stage piloted vehicle mass, IMEO = 754.7 t
(Isp = 480 s) 2-Stage cargo vehicle mass, IMEO = 800.2 t

* Two 250-klbf-class NTR's are used in the TMI stage.

Note: Above referenced LESC nos. are preliminary estimates which will be substantiated in subsequent study analysis.

than its aerobraked chemical counterpart. It should be noted, however, that the LESC results are based on a scale-up of vehicle masses required for a 2004 piloted opportunity. It is possible to show a mass savings for the piloted vehicle by eliminating the cooldown propellant for the expendable TMI stage and increasing the Isp to 950 s.

Conclusions and Recommendations. "Old" SC/NTR technology can provide "new" high-leverage capability for human expeditions to Phobos and Mars. For the all-propulsive split mission to Phobos, reductions in IMEO on the order of 40 to 50 percent appear possible. For split missions to Mars, the NTR (operating all propulsively) can still provide a 5 to 15 percent savings in IMEO over that of the aerobraked chemical system. With comparable propellant loadings, the SC/NTR could travel faster, higher delta V transfer orbits than its chemical counterpart, resulting in further reductions in crew trip time. Similarities to previously studied NTR concepts suggest that parameters be defined for a modular NTR stage capable of performing a variety of missions. By appropriately sizing the engine, a single NTR stage could function as a lunar shuttle; by clustering, several NTR stages could be used to support human expeditions to Phobos and Mars.

Planned FY 1989 Trade Studies.

a. Obtain improved estimates of shielding and cooldown propellant requirements for NTR systems.
b. Determine the appropriate engine size(s) and initial thrust-to-weight for current case studies of interest (this effort will involve tradeoffs of gravity losses, engine weight, and operating life).
c. Study the sensitivity of IMEO to variations in hyperbolic excess velocity for SC/NTR systems with differing values of thrust and specific impulse.
d. Identify the implications and requirements of operating a reusable nuclear stage or stages.
e. Examine the safety issues associated with operating NTR's near piloted facilities or on a piloted spacecraft.

5.2.3. Impact of Phased Implementation of Solid and Gas Core Nuclear Thermal Rocket (GC/NTR) Propulsion on the Lunar Outpost to Early Mars Evolution Case Study

The objective of this trade study was to estimate the savings in IMEO, and the logistical simplifications resulting from the introduction of increasingly more efficient GC/NTR technology into the Lunar Outpost to Early Mars Evolution case study.

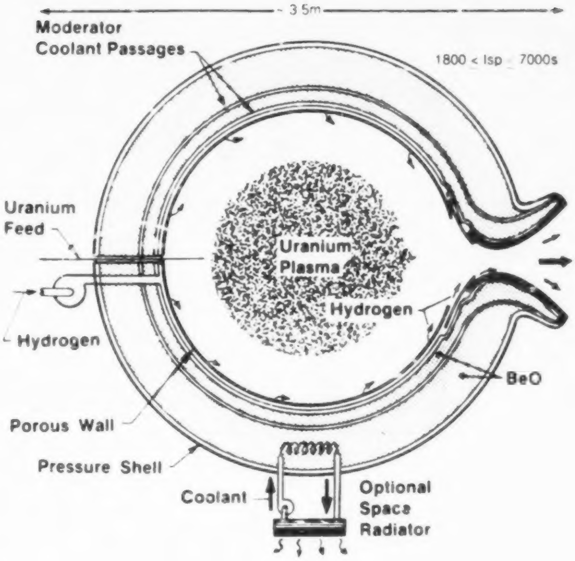
~ 3.5m
Moderator
Coolant Passages
1800 < Isp < 7000s
Uranium
Feed
Uranium
Plasma
Hydrogen
Hydrogen
BeO
Porous Wall
Pressure Shell
Coolant
Optional
Space
Radiator

Figure 5.2.3-1.- High specific impulse, porous wall gas core engine.

Background. During the Rover/Nerva program both solid and gaseous core rocket concepts were studied. The solid core (SC)/NTR was considered by NASA to be the logical first step toward achieving a working engine. With a specific impulse capability of ~850 to 1000 s, the high thrust SC/NTR could perform a variety of lunar and interplanetary missions, with lower IMEO than chemical systems and shorter trip times than with electric propulsion. With comparable propellant loadings, the SC/NTR could also travel faster, higher delta V transfer orbits than its chemical counterpart, resulting in reduced mission times. Beyond the SC/NTR, NASA plans called for the development of advanced gaseous core (GC)/NTR engines that would operate over an Isp range of ~1500 to 7000 s. With high thrust (F) and Isp capability, the GC/NTR would make quick round-trip missions to the nearby planets a real possibility. To achieve this level of performance, a high temperature (~10^4 to 10^5K) cylinder or sphere of fissioning uranium plasma would serve as the fuel element, thereby eliminating the material temperature limitations of the SC/NTR. In the gas core concept, nuclear heat released within the plasma is dissipated as thermal radiation and is absorbed by a surrounding envelope of hydrogen propellant, which is then expanded through a nozzle to provide thrust. Two concepts studied in significant detail during the Nerva program showed considerable promise: an open-cycle configuration (figure 5.2.3-1), and a closed-cycle approach, known as the "nuclear light bulb" (NLB) engine (figure 5.2.3-2). The NLB offers the potential for perfect containment of both

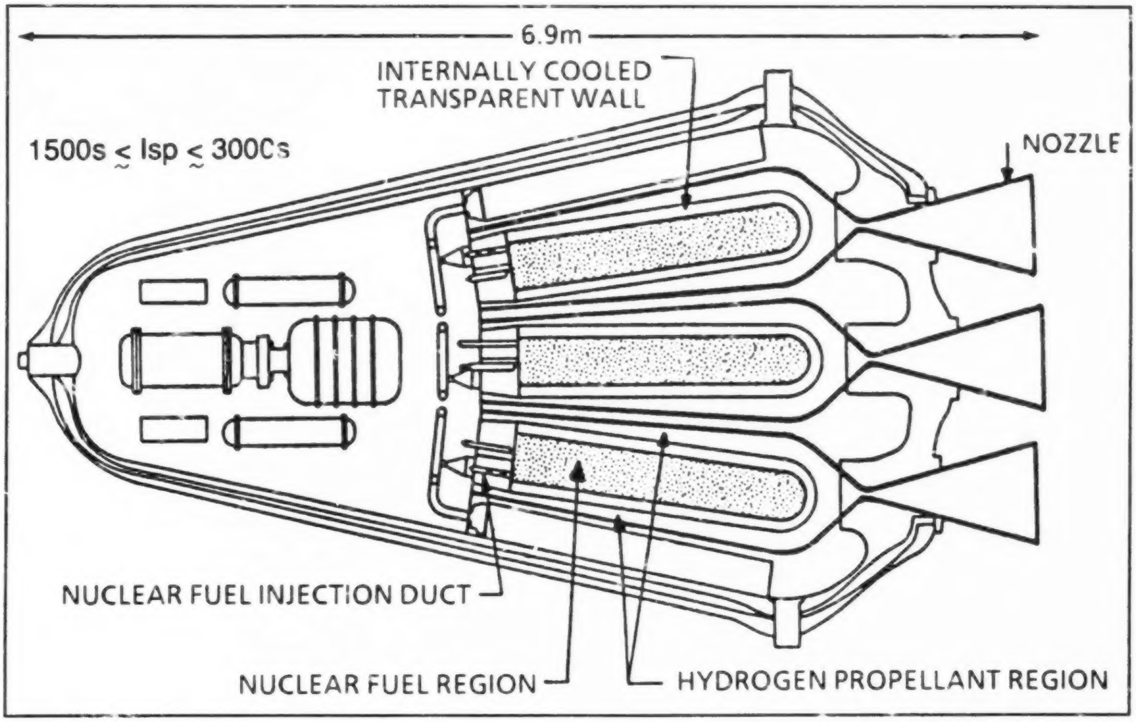
6.9m
INTERNALLY COOLED
TRANSPARENT WALL
1500s ≤ Isp ≤ 3000s
NOZZLE
NUCLEAR FUEL INJECTION DUCT
NUCLEAR FUEL REGION
HYDROGEN PROPELLANT REGION

Figure 5.2.3-2.- Sketch of the nuclear light bulb engine.

fuel and fission products through the use of a cooled transparent wall structure. The porous-wall, open-cycle system uses fluid mechanical means for separating the gaseous fuel and propellant. While a small percentage (<1%) of fuel is exhausted along with the propellant, the absence of an internal moderator in the open-cycle concept allows the radiator-cooled version of this engine to operate at significantly higher values of Isp—in the range of 3000 to 7000 s.

Trade Studies.

Trade No.	Description
1	Baseline — Quantify the mass savings for the cargo andpiloted sprint missions to Mars (2009 and 2011 opportunities) obtained using closed-cycle, NLB gas core technology.
2	Show the sensitivity of IMEO vs. round-trip time for "all-up" exploration class missions to Mars using high performance, space-radiator-cooled, open-cycle gas core rocket (SRGCR) technology.

Key Assumptions.

a. Closed-cycle NLB and open-cycle SRGCR engines are assumed available around the 2010 timeframe

b. NLB parameters: $F = 10^5$ lbf, $Isp = 1870$ s, $M_{eng} \approx 38$ t (includes 4.5-t external disk shield)

c. SRGCR parameters: $F = 5 \times 10^4$ lbf, $Isp = 5700$ s, $M_{eng} \approx 120$ t (includes a 60-t space radiator)

d. GC/NTR stage mass (not including engine) ranges from 16% (NLB) to 20% (SRGCR) of the total propellant load

e. "All-propulsive" braking (no aerobraking) is the only option considered. In trade study no. 1, a two-stage piloted vehicle and single-stage cargo vehicle are assumed. After injecting the piloted stage, the reusable TMI stage contains sufficient propellant for a retro maneuver and Earth orbital return to space station (500 km).

f. In trade study no. 2, an "all-up" vehicle carrying both crew and cargo is assumed. The cargo manifest includes 150 t left at Mars and 100 t returned to Earth.

Analytical Approach. Both trade studies use delta V budgets that consistently take into account mission C_3 requirements, the spacecraft thrust-to-weight (F/Wi), and gravity losses. In trade study no. 1, the C_3 criteria for the piloted mission with aerocapture (w/AC) and the cargo mission without aerocapture are 75 and 10.3 km^2/s^2, respectively. The piloted vehicle uses the reference delta V budget[1] for the deep-space andMars escape maneuvers, and approximately the same sequence of maneuvers for achieving the reference Mars parking orbit. Braking, both at Mars and at Earth return, is done all-propulsively. In trade study no. 2, C_3 requirements consistent with a range of round-trip travel times are used, and gravity losses are taken into consideration.

To obtain IMEO estimates for the cargo, sprint, and "all-up" class spacecraft, the propellant loading must be determined. Using the specified payloads and GC/NTR scaling assumptions, and knowing the time history of the vehicle mass, the propellant requirements are estimated using the Rocket Equation.

Findings. The two trade studies yielded the findings summarized below.

Trade #1. For the same trip times, the use of closed-cycle NLB technology allows the reference cargo/sprint missions to be performed "all-propulsively" with less launch mass in Earth orbit (500 km) than that required by the NEP cargo and aerobraked chemical systems launched from lunar orbit. The mass reduction is ~414 t, which represents a savings of ~21% (table 5.2.3-I). The logistical complexity of the mission and of lunar base operations is also reduced. In the reference case study, six vehicles/stages are utilized. Included in this inventory are 1 NEP cargo vehicle, 1 NEP lunar orbital transfer vehicle (OTV), 1 piloted vehicle (w/AC), 2 trans-Mars injection (TMI) stages (w/AC), and 1 chemical lunar OTV (w/AC) used to transport the crew from Space Station Freedom to the piloted vehicle departing lunar orbit for Mars. A total[2] of ~840t of LLOX must also be produced to fuel the logistics landers, excursion modules, and piloted vehicle stages used during the Mars mission. With NLB technology, a single stage cargo vehicle and a two-stage piloted vehicle are all that is required. Because the NLB requires only LH_2 propellant, the infrastructure for producing, storing, and ferrying up ~840 t of LLOX is unnecesssary, and can be used to support other lunar base activities.

Trade #2. SRGCR's can perform "all-up," all-propulsive exploration-class missions to Mars in ~280 days (including a 40-day stay at Mars) with an IMEO of ~1000 t (figure 5.2.3-3). Increasing the mission time to ~450 days (the duration of the split-mission sprint leg) lowers the IMEO to ~600 t. With the SRGCR, the cargo leg of the split option is unnecessary, and significant reductions in both mass (from 2018 t down to 1000 t) and number of required vehicles/stages (from 6 down to 1) are possible (figure 5.2.3-4).

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[1] Trajectory data and the delta V requirements for major space maneuvers provided by SAIC.

[2] Mars mass summaries for the NEP cargo and piloted sprint missions provided by LESC

TABLE 5.2.3-I.- EVOLUTIONARY MISSION [1] RESULTS USING NLB TECHNOLOGY

Parameters	Piloted Chemical[2] (Isp=480 s)	Piloted NLB (Isp=1870 s)	Cargo NEP[2] (Isp=6000 s)	Cargo NLB (Isp=1870 s)
• Piloted vehicle mass at launch, t	639	286.2	—	—
• Reusable injection stage mass at launch, t	590 (2 stages)	587.4 (1 stage)	—	—
• Piloted vehicle stack mass at launch, t	1229	873.6	—	—
at LEO, t	715	873.6	—	—
• Cargo vehicle stack mass at launch, t	—	—	789	730.1
at LEO, t	—	—	656	730.1 (1 stage)
• Piloted vehicle mass savings[3], t	—	355.4	—	—
• Cargo vehicle mass savings[3], t	—	—	—	58.9
• Total Mass Savings[3], t	414.3			

[1] Results are for the 2011 sprint opportunity (ΔV data from SAIC)
[2] Mass data provided by LESC
[3] Savings based on launch mass comparison

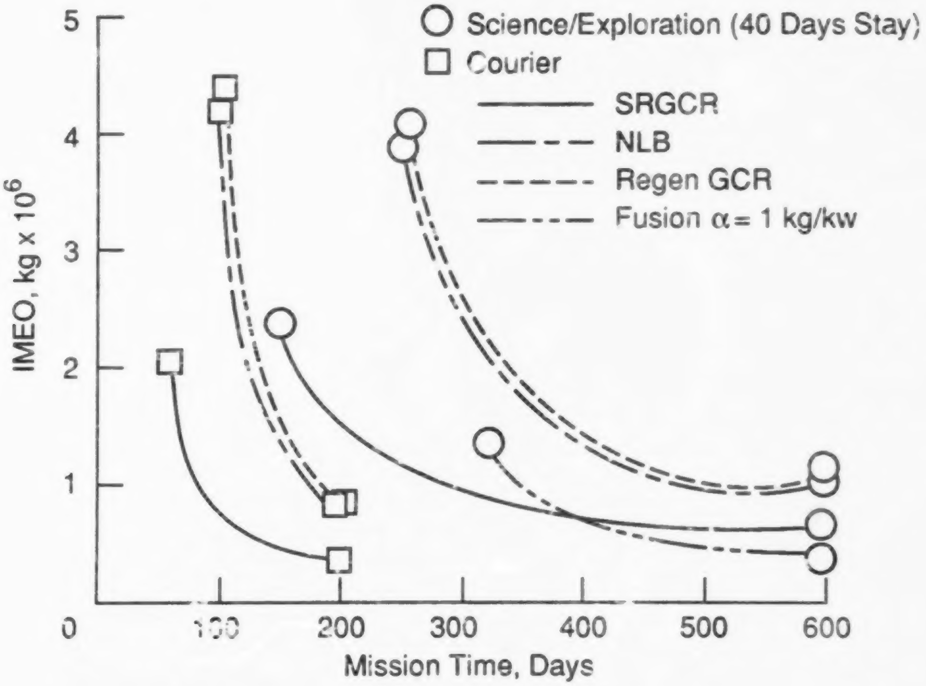


Figure 5.2.3-3.- Sensitivity of IMEO with Mars round-trip mission time.

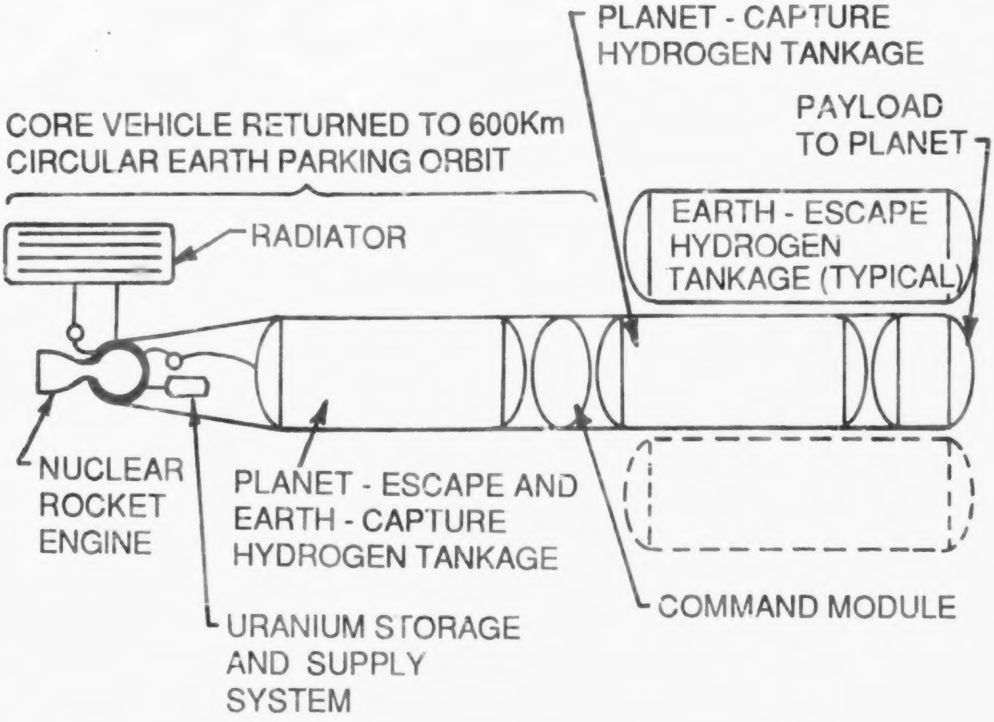

PLANET - CAPTURE HYDROGEN TANKAGE
CORE VEHICLE RETURNED TO 600Km CIRCULAR EARTH PARKING ORBIT
PAYLOAD TO PLANET
RADIATOR
EARTH - ESCAPE HYDROGEN TANKAGE (TYPICAL)
NUCLEAR ROCKET ENGINE
PLANET - ESCAPE AND EARTH - CAPTURE HYDROGEN TANKAGE
COMMAND MODULE
URANIUM STORAGE AND SUPPLY SYSTEM

Figure 5.2.3-4.- Schematic of reusable, radiator-cooled GCR spacecraft.

Conclusions and Recommendations. GC/NTR technology offers the potential for providing high-thrust (~10^5 lbf)/high-Isp (1500 to 7000 s) propulsion systems, which could make convenient transportation of crew and cargo between Earth and Mars a reality. In addition to a significant analytical effort, the GCR research program successfully demonstrated key functional aspects of both GCR concepts in a variety of non-nuclear experiments (some using RF-heated uranium plasma sources to simulate engine operation). Plans for important nuclear tests at both the sub- and full-scale engine levels were also under development at the time the Nerva program was terminated.

The above results suggest that GC/NTR's could be available around the 2010 timeframe. Because of the high leverage this technology appears to offer, it is recommended that GC/NTR's be considered in future evolutionary case studies.

Planned FY 1989 Trade Studies.

a. Develop improved scaling relationships for estimating the mass of a gas core rocket (GCR) stage.

b. Study the implications/benefits of using GCR's for cislunar space transportation (e.g., with its high Isp, the GCR could carry larger payloads or enable quick courier trips (of 24 hrs or less) to the Moon).

c. Study sensitivity of IMEO to variations in hyperbolic excess velocity for GCR's with differing values of thrust/specific impulse.

d. Determine the optimum thrust/Isp allowing lowest IMEO and round-trip travel times to Mars.

5.2.4 Issues of Mars Orbital Refueling

The objective of this study was to identify and clarify the options and issues associated with refueling a piloted vehicle (PV) in Mars orbit, assuming a cargo vehicle (CV) with propellants has preceded the PV into Mars orbit. Other potentially viable Mars orbit operations that might be used in lieu of refueling (e.g., propellant tank or crew module transfers) were also identified and initially analyzed. The trade space considered four locations for Mars orbit operations, four refueling/transfer options, and a dozen issue areas.

Background. Current concepts for human expeditionary missions to Phobos and Mars surface employ "split" missions, in which cargo vehicles containing the propellants for Earth return are captured in Mars orbit prior to departure of the crew from LEO. Thus, the PV must rendezvous with the CV in Mars orbit and perform critical transfer operations to achieve mission success. These scenarios will require a detailed study of the issues identified and illuminated in this study. This study was performed by Anlex Corporation and General Dynamics Space Systems (GDSS) Division.

Key Assumptions.

a. Expeditionary missions to Phobos and Mars surface are used as reference cases.

b. Zero-g cryogenic propellant transfer capability will be developed for a multiplicity of space applications, including Mars orbit refueling (MOR).

c. Shuttle and Space Station Freedom rendezvous and docking experience will confirm high crew proficiencies and provide training opportunities.

d. There will be a continuity of repair and maintenance philosophies from Shuttle through Space Station Freedom and Mars expeditions.

e. There will be a lack of training opportunities in milli-g environments prior to the expeditionary missions.

f. Remote manipulator system (RMS) technology can be scaled from the current 32,000-lb capability to a 250,000-lb capability.

Approach. Data on the physical environments in Mars orbit and on/near Phobos and Deimos were compiled and assessed. Our studies of orbital transfer vehicle (OTV) refueling/servicing operations and long-term cryogenic storage facilities—both in LEO—were reviewed. This large data base was then extrapolated to the Mars environment, where issues and options associated with MOR were identified and subjected to preliminary analysis.

Findings. Twelve issues of importance to refueling and/or transfer operations in Mars orbit were identified in this study:

a. Man-hours, including EVA and IVA (using the RMS)

b. Total refueling/transfer operations versus the total time near Mars

c. Power demands

d. Operational flexibility and crew safety

e. Compatibility of refueling/transfer operations with overall mission goals

f. Storage environment, particularly regarding thermal management of cryogenics

g. Technology requirements

h. Orbital rendezvous options, including optimal CV orbits for minimum-capture delta V, recovery/tracking, and PV rendezvous

i. Proximity operations, including possible Phobos/Deimos docking

j. Crew training in free space versus the Phobos/Deimos environment

k. Precursor mission and system requirements

l. Programmatic development capability

Table 5.2.4-I summarizes the options and issues associated with the four location options: low Mars orbit (LMO), high Mars orbit (HMO), on Phobos, or on Deimos.

Highly elliptical, inclined orbits have the lowest CV/PV capture delta V's and allow economical plane changes; they also impose only moderate Mars thermal loads on the CV. Although docking the CV with Phobos or Deimos would remove uncertainties in its orbital position, the thermal management advantages inherent in shadowing from Mars may be minimal due to heat transfer from the "warm" moons themselves. This scenario would also require an automated CV landing on Phobos or Deimos. Refueling and/or transfer operations (summarized in table 5.2.4-II) can occur in free space (if desired), using whatever techniques (and gravity levels) are developed in LEO. The milli-g environment on Phobos/Deimos could be used as a backup (e.g., for fluid acquisition) or for crew training (in anticipation of in-situ propellant production, which will require milli-g operations); Phobos/Deimos gravity, topography, and dust environments would have to be specified and accommodated by the operations and vehicles.

Table 5.2.4-II summarizes our findings concerning the four most feasible transfer options in Mars orbit:

a. Refueling the PV with CV propellants (i.e., fluid transfer)

b. Transfer of CV propellant tanks to the PV

c. Transfer of the crew module from the PV to the CV

d. Transfer of the crew from the PV to the CV (i.e., the PV and CV are essentially identical)

The tank transfer technique requires three to four times the total operational time of the other transfer options, assuming 7 to 21 kW sources are available (to reduce the fluid transfer time). In addition, its operational complexity and long list of other negatives makes tank transfer the least preferred technique. Although crew module transfer appears to be a viable method, its operational advantages (if any) depend on the details of the currently undefined crew module interface. The simplest, quickest option is complete PV and CV redundancy. However, this option may be the most massive and most costly and requires thorough checkout of the CV before Earth return. Having PV/CV redundancy would provide an excellent contingency option. Refueling the PV is a simple operation that apparently can be completed in a

TABLE 5.2.4-I.- LOCATION OPTIONS SUMMARY

	Low Mars Orbit	**High Mars Orbit**		**On Phobos**	**On Deimos**
	Circular	Circ/equatorial	Elliptic/inclined		
CV & PV					
Capture orbit	High capture ΔV	Medium cap. ΔV low ΔV for transit to PH/D	Lowest cap. ΔV low ΔV plane change at apoapsis	Medium capture ΔV	
CV					
Thermal management	High thermal load from Mars surface (store prop. as H2O.then electrolyze?	Low thermal load from Mars	Moderate thermal load from Mars	Shielding from Mars but surface heat conduction an issue	
Locating/ tracking	Equat.-high rate of nodal regression Inc.-lower rate of nodal regression	Slower orbit changes	High e, low i makes nodal/apsides shifts significant for tracking problem	Best Option for Locating CV Unmanned rendezvous complicated by complex Phobos & Mars gravity field	Unmanned rendezvous less complex for Deimos
PV					
Rendezvous/ docking	Rendezvous more complex than for higher orbits due to nodal regression	Rendezvous somewhat simpler than for LMO	Rendezvous somewhat simpler than for LMO	Manned rendezvous with moons possibly more reliable than for unmanned CV since PV under real-time control	
Refueling transfer environment	Free space (use LEO methods)	Free space (use LEO methods), possible dust belt environment	Free space (use LEO methods)	Milli-g (free space avail), dust environment	
Mars landing	Lower ΔV for landing and ascent	Higher ΔV	Higher ΔV	Same class as HMO/circ. - Deimos higher ΔV than Phobos	
Mars escape	Higher ΔV	Lower ΔV	Lower ΔV	Same class as HMO/circ. - Deimos lower ΔV than Phobos	

TABLE 5.2.4-II. TRANSFER OPTIONS SUMMARY

	Fluid Transfer		**Tank Transfer**	**Crew Module Transfer**	**Redundant vehicles**
Minimum time	40 hrs	5 hrs	16 hrs (?)	4 hrs (?)	2 hrs (?)
Power	0.9 - 2.6 kWe	7-21kWe	TBD	Minimal	N/A
Operational complexity	Single fluid interface possible		16 interfaces to break + make	Crew module interface unidentified. simple(?)	Simplest option
Pros (+) and cons (-)	+ Simple operation +Crew returns in known vehicle + Not crew intensive - Additional power may be needed		+ Possible low power (TBD) - Crew intensive - Complex ops - Slosh problems - Greatest demands on RMS	+ No additional power required + Potentially simple ops (?) - Constrains vehicle design - Return vehicle needs checkout	+ Quickest(?) + Offers best contingency + Not crew intensive + Insensitive to PH/D plumbline - Most massive - Return vehicle needs checkout
conclusion	viable method alone and valuable backup		least preferred	possibly viable	viable method

	Phobos	**Deimos**
Orbit		
Elements : a,e,i	2.77 Rm, 0.015, 1.02° **Deeper in Mars gravity well**	6.92 Rm, 0.00052, 1.82° **Easier access from high parking orbits** **More solar power availability** **Easier access to high Mars latitudes** **Better communication with Earth**
Sidereal Period, Rotation	7h 39m, Synchronous	30h 18m, Synchronous **More continuous monitoring of Mars**
Orbital velocity of moon/orbital dust Belts	2.1 km/s / **Belts probable**	1.4 km/s / **Belts possible**
Physical		
Mass, density, Major dimension	9.8×10^{18} g, 2.0±0.5 g/cc, 27km	2.0×10^{18} g, 1.9±0.7 g/cc,15km
	Densities consistent with type I or II carbonaceous chondrites	
Surface gravity (cm/s2)	Max (90°N&S Lat): 0.73 Min(Sub/Anti Mars Pts):0.32	0.36 (Ref.:Davis et al (1981), 0.30 Icarus 41,p.220)
Escape velocity variations	3.5 - 15.5 m/s **Complex local gravity field**	6 - 7 m/s **More uniform gravity field**
Surface		
Albedo	5%, Homogeneous	6%, Bright areas
Photometry, Polarization, Thermal	Lunar-like regolith (100m Thick)	Lunar-like Regolith (20m Thick)
Spectra	**Similar to type I or II carbonaceous Chondrites (0.2-0.6 μm)**	
Craters	Largest crater: 10km Few craters regolith filled; Layering visible	Largest crater: 3km Most craters regolith filled; No layering
	Crater densities close to lunar highlands	
Morphology	Grooves related to Stickney **Possible evidence for H2O**	No grooves; downslope Motion of regolith; retained most of crater Ejecta **Relatively unmodified body**

Figure 5.2.4-1.- Phobos & Deimos: characteristics & discriminators

short time. Fluid transfer could also serve as a valuable backup plan for any of the other options.

A summary of physical characteristics of Phobos and Deimos and their associated operational discriminators is shown in figure 5.2.4-1. Deimos' larger orbital semi-major axis has advantages with respect to high orbit access, high Mars latitude access, solar power, Earth communications, and continuous Mars surface monitoring. Available data on albedos, densities, and spectra suggest that both objects are probably composed of significant amounts of water (about 10 percent); it is likely that Deimos is less affected by impacts. Deimos' smoother local gravity field is less challenging for proximity operations than Phobos' complex gravity.

Table 5.2.4-III recaps our conclusions and recommendations. More discussion of the above topics is available in the Appendix.

5.2.5 Launch Of Cryogenic Tanks

A brief study was performed to determine the better method of launching cryogenic storage tanks and their

TABLE 5.2.4-III. - PRELIMINARY TECHNICAL RECOMMENDATIONS

Refueling in Mars orbit greatly preferred over all-up two-way chemical sprint vehicle
Preferred storage/transfer location: • High mars orbit • Deimos surface as contingent location
Preferred transfer options: • Fluid transfer, or • Redundant vehicles with fluid transfer capability
Technology development (if expeditionary missions become drivers): • Accelerate cryogenic fluid transfer technology development • Develop technology for remote system status monitoring
Training opportunity creation: • Structure Mars precursor mission(s) to use large-scale LEO refueling, even if the mission(s) would not demand it • Strive for crew continuity from above precursor(s) to Mars expedition

contents to LEO, by using either full storage tanks or separate transfer tanks. The study was based on the fact that cryogenic storage tanks designed for launch with propellant must have heavier structure and therefore increase mission dry mass.

Methodology. A simple approach was used to determine a first-order impact on launching cryogenics in tankage designed to withstand launch loads with a full complement of fuel versus launching a flight with the tank empty and transferring propellant from a dewar in space. The approach concentrates on weight and thermal impacts and on technology advances in thermal protection that will lower boiloff. No attempt was made at this time to determine the impacts from an operational standpoint, such as debris shielding in LEO during assembly of the vehicle.

Key Assumptions.

a. Conventional chemical propulsion systems (LOX/LH_2) with a mixture ratio 6:1 engine

b. Assembly time of 1 year

c. Tankage mass fractions (shown in figure 5.2.5-1) from "Long-Term Cryogenic Storage Facility Systems Study," (reference 1) MSFC contract NAS8-36612 overview presentation to LeRC, October 1987

d. A 10 percent propellant loss due to fluid transfer

e. Increases in boiloff for tanks launched full (mostly due to larger and more numerous struts) versus launching empty tanks: 60 percent for LOX and 30 percent for LH_2 (from Aydalott, John C. "Thermal Analysis of Space Station Cryogenic Propellant Depot Concepts." LeRC paper). See figures 5.2.5-2 and -3.

f. Technology levels as follows:
   (1) State-of-the-art, 60 layers multilayer insulation (MLI), fiberglass/epoxy struts
   (2) Minimal technology advance, 120 layers MLI, vapor-cooled shield
   (3) Moderate technology advance, 120 MLI, decoupled struts, etc.
   (4) Significant technology advance, add refrigeration to Level 3

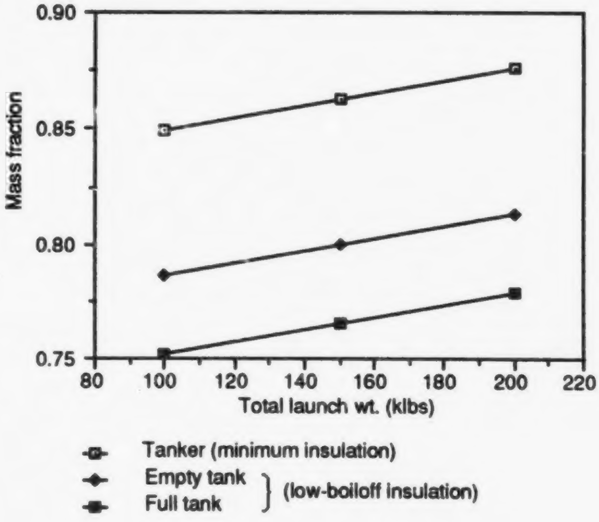


Figure 5.2.5-1.- Mass fraction of tanks.

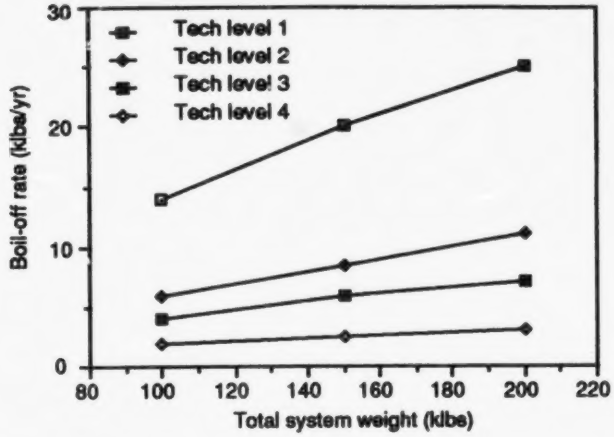


Figure 5.2.5-2.- Boiloff rates for hydrogen tank.

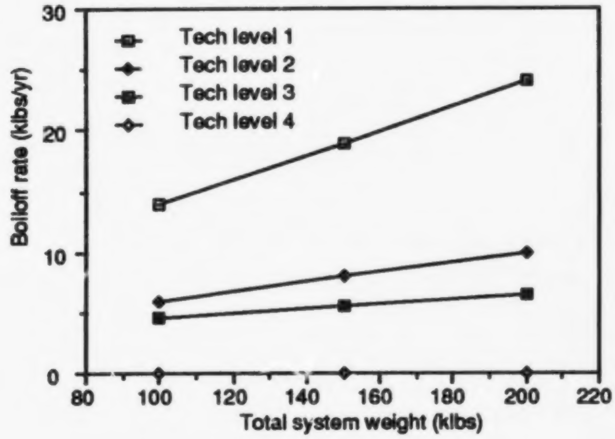
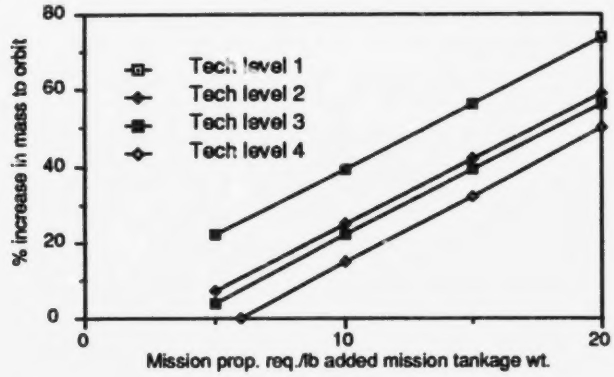


Figure 5.2.5-3.- Boiloff rates for oxygen tank.



%Increase = boiloff diff + tank tare wt. impact - tanker wt. - transfer losses

Figure 5.2.5-4.- Percent increases in propellant to LEO above fluid transfer option.

Findings. The launching of propellant to LEO in the flight tankage will have a severe penalty in additional propellant (and tankage) required to accommodate the excess tankage weight and boiloff for a Mars mission. The penalty could be as severe as 50 percent for state-of-the-art technology to 15 percent with a significant technology advance over a fluid transfer option, as shown in figure 5.2.5-4. Due to the high boiloff rates with state-of-the-art technology, a moderate technology advance will be required for a manned Mars mission of any significance that uses LOX/LH2 for propellant. A lunar mission would not have such a severe penalty and would have to be investigated in depth to determine the most appropriate option.

5.3 ADVANCED LIFE SUPPORT SYSTEMS

5.3.1 Life Support Architecture and Technology Requirements Definition

The objective of this assessment was to conceptualize a life support system (LSS) approach including requirements and technology options to accommodate the Office of Exploration (OEXP) lunar and Mars mission case studies. This activity includes an assessment of existing technology limitations and their effects upon LSS requirements and mission objectives.

Background. The advanced-mission case studies under consideration by OEXP may require unique life support technology differing from that used for Shuttle or that planned for the Space Station Freedom. The life support requirements for Shuttle are satisfied by using open-loop approaches (i.e., stored oxygen, water, and supplies, with wastes returned to Earth). The existing technology available for Shuttle is applicable only for small crew sizes and short mission durations. Present space station planning includes technology for partial recovery of consumables by physicochemical means, enabling a partially closed life support system that will reduce operational costs associated with Earth-based resupply.

However, future manned planetary missions generate requirements fundamentally different from those which drove the design of the Shuttle life support system and are presently shaping the technology planned for Space Station Freedom. Most significant of these unique mission differences are the longer mission durations, the larger crew size, the inability to resupply expendables quickly, and the high reliability required due to potentially long mission abort times. This unique combination of requirements dictates a life support system that will be characterized by a higher degree of closure, high reliability, increased automation, and independence from terrestrial resources.

Key Assumptions/Ground Rules. The LSS includes the following functions:

a. Atmosphere revitalization: habitat pressure and composition control, oxygen and diluent gas supply, carbon dioxide and trace contaminant removal

b. Temperature and humidity control: habitat heating, cooling, dehumidification, and ventilation

c. Food as it influences the LSS, including preparation, storage, metabolic balance, and biological provisions

d. Personal hygiene: provisions for handwash, shower, urinal/commode, and laundry-dishwasher when needed

e. Water management: ensured quantity and quality of water for drinking, food preparation, clothes- and dishwashing, and bathing

f. Waste management: urine, feces, and trash disposal or recycling

The LSS technology used for Shuttle Orbiter and baselined for the initial Space Station Freedom will be considered a starting point for future mission analyses and trade studies.

Approach. The OEXP advanced mission case studies will be evaluated in detail to determine how life support is influenced by mission characteristics and interfacing system considerations. This evaluation will be a continuing process as missions become better defined. Top-level requirements and an LSS conceptual approach will be determined for various mission options to meet mission goals and physiological needs. From the top-level LSS approach, various options can be evaluated (e.g., closed ecological life support system (CELSS) payoff point, alternate technology potential, and utilization of in situ materials).

Findings. A preliminary study was conducted in late FY 1988 to evaluate the use of baseline space station regenerable and Shuttle open-loop LSS technology in the OEXP mission case studies. The study examined only the basic life support functions of carbon dioxide control, metabolic oxygen generation, potable and hygiene water recovery, waste management, and provision for food and clothing. No attempt was made to estimate a complete LSS ship set including system redundancy, gas supply for atmosphere makeup or repressurization, ducts and plumbing, or support items such as refrigerator and galley provisions. Shuttle open-loop technology was used for the surface portion of the humans-to-Mars case and for lunar observatories due to the small crews and relatively short staytimes. Space Station Freedom regenerable technology was applied to the Phobos, Humans-to-Mars transport, and lunar-outpost-to-Mars. Biological life support was not included because of the small crew sizes for the case study missions and the insufficiency of the data base to adequately represent this technology.

The results in table 5.3.1-I show that, while baseline Space Station Freedom and Shuttle technology are technically feasible, a significant penalty largely associated with food and water is incurred by their use in the case studies. For example, the humans-to-Phobos mission requires 9,268 lb (4203 kg) of makeup water because of process inefficiencies.

Planned FY 1989 Activity. The Space Station Freedom and Shuttle technology trade study will be refined and expanded to more completely identify process limitations and to identify areas which impose the greatest mission impacts.

A new program was initiated at the end of FY 1988 to conceptualize a life support approach for the OEXP mission case studies. As shown in the abbreviated task flow of figure 5.3.1-1, the program is divided into two phases. Phase 1 to be accomplished during FY 1989, will generate life support requirements based on detailed case study mission characteristics and potential interfacing system influences.

This task involves indepth study of the four case studies with emphasis on possible variations within each mission which will influence the LSS. The study will use a data base derived from an exhaustive literature survey, the key parameters that fundamentally drive the individual missions, and the interrelationship with other systems, such as power and thermal control. Mission characteristics for each of the case studies will be defined by such parameters as overall mission purpose, environmental conditions (gravity, atmosphere, and radiation), mission timeline (launch opportunities, length, significant events, extravehicular activity (EVA) events, and resupply periods), crew size, and abort options. Based on these mission characteristics, top-level LSS requirements for the OEXP mission case studies will be established and satisfied using simplified LSS options (e.g., Space Station Freedom technology, in situ O_2 production, and waste closure). Once this preliminary selection process is complete, LSS option refinements will be studied to determine their effects on the mission alternatives. This step focuses life support requirements by modifying options such as degree of closure, use of extra-terrestrial resources, and physicochemical versus biological approaches.

Phase 2, which will be completed in FY 1990, will refine the basic LSS used for each of the mission cases. An understanding of the base/spacecraft LSS requirements developed during Phase 1 will assist with identification of technologies that are compatible with the rest of the vehicle. This data base will result in system level configurations for each mission case study. LSS conceptual

TABLE 5.3.1-I.- FY88 LIFE SUPPORT SYSTEM STUDY RESULTS:
USE OF SPACE STATION FREEDOM AND SHUTTLE TECHNOLOGIES TO
ACCOMPLISH MISSION CASE STUDIES

Mission (Technology)	Weight (kg/lbs) per flt.					Launch volume (ft^3/flt)	Power (kW)
	Systems 1	System Spares 2	Consum-ables 3	Water makeup 4	Total launch		
Human expedition to Phobos (station)	964/ 2125	277/ 610	3097/ 6829	4203/ 9268	8541/ 18833	1034	4
Human expeditions to Mars							
Transport (station)	1927/ 4251	553/ 1220	6195/ 13659	8406/ 18535	17081/ 37665	2069	8
Surface (Shuttle)	574/ 1265	3/6	1955/ 4312		2532/ 5583	250	1.3
Lunar observatory (Shuttle)	574/ 1265	3/6	1955/ 4312		2532/ 5583	250	1.3
Lunar outpost to early Mars evolution							
Lunar portion (station)	964/ 2125	240/ 530	2692/ 5935	3652/ 8054	7548/ 16644	922	4
Mars portion (station)	1928/ 4251	1778/ 3920	19911/ 43904	27019/ 59578	50636/ 111653	5869	8

Notes:
1 Single subsystem only: (no redundancy, tankage, atmosphere makeup, plumbing, etc.) to accomplish functions of air revitalization, water and waste management - not a complete LSS ship set.
2 Spares estimate to maintain subsystem operation for mission duration.
3 Includes food and consumables necessary to maintain system operation.
4 Includes water and tankage to make up for subsystem inefficiency.

configuration data for near-term development and LSS technology needing additional development will be defined and documented.

5.3.2 Extravehicular Activity System Requirements Definition

This study will identify technology requirements for the extravehicular activity system (EVAS) necessary to complement future lunar and Mars missions.

Background. EVA capability will be necessary to accomplish the goals of future manned missions to the Moon and Mars. The extent of this capability could vary widely, depending upon specific mission objectives. In synergism with the previous Space Station Freedom and geosynchronous orbit EVAS definition studies, this work will examine the OEXP case study missions in detail to determine EVAS requirements. Results will provide planners with data to define the extravehicular portion of future missions and to identify necessary EVAS technology requirements to support those activities.

Key Assumptions/Ground Rules.

a. The study will examine and consider the entire extravehicular activity operation, including suits, vehicles, special equipment, and airlocks. This broad assessment is necessary to ensure compatibility between equipment and tasks to be accomplished.

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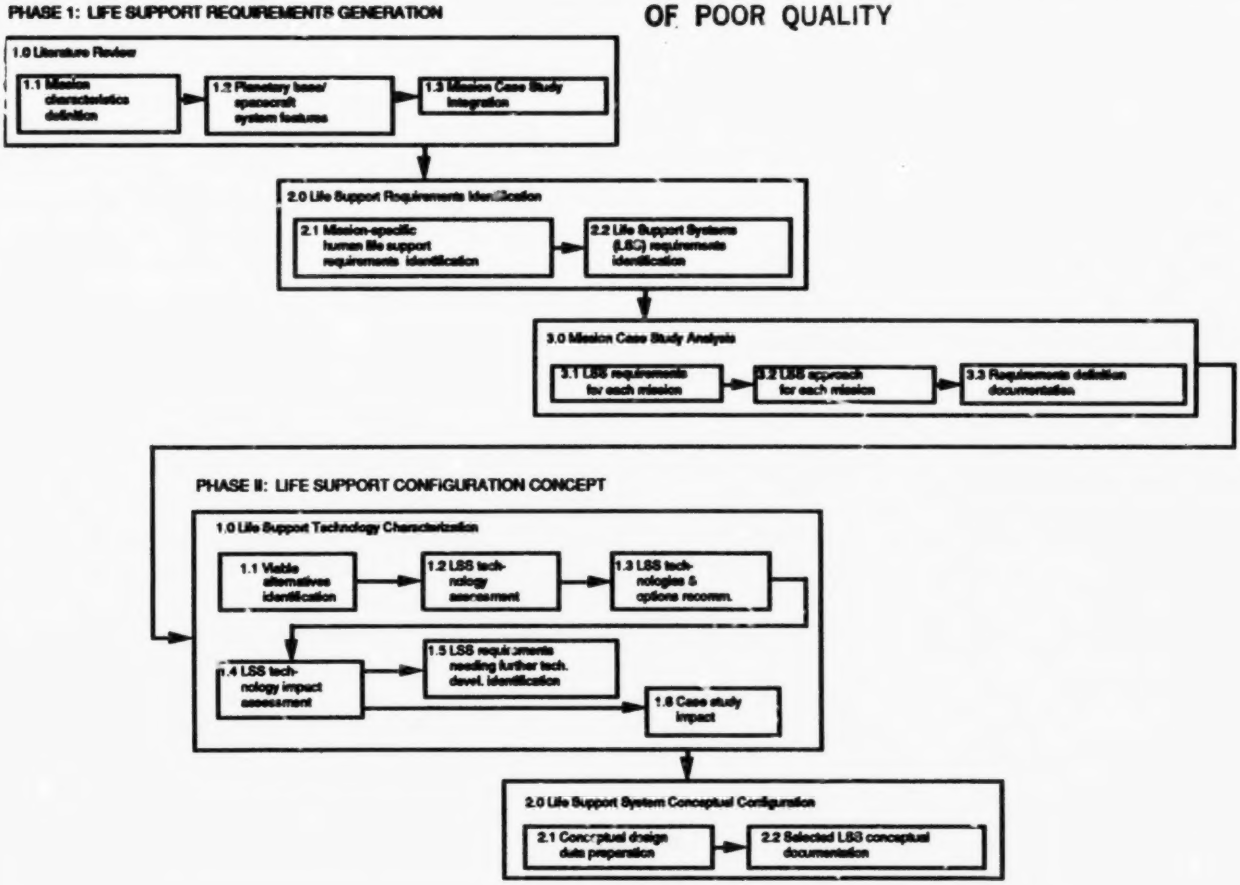
PHASE 1: LIFE SUPPORT REQUIREMENTS GENERATION
1.0 Literature Review
1.1 Mission characteristics definition
1.2 Planetary base/ spacecraft system features
1.3 Mission Case Study Integration
2.0 Life Support Requirements Identification
2.1 Mission-specific human life support requirements identification
2.2 Life Support Systems (LSS) requirements identification
3.0 Mission Case Study Analysis
3.1 LSS requirements for each mission
3.2 LSS approach for each mission
3.3 Requirements definition documentation
PHASE II: LIFE SUPPORT CONFIGURATION CONCEPT
1.0 Life Support Technology Characterization
1.1 Viable alternatives identification
1.2 LSS technology assessment
1.3 LSS technologies & options recomm.
1.4 LSS technology impact assessment
1.5 LSS requirements needing further tech. devel. identification
1.6 Case study impact
2.0 Life Support System Conceptual Configuration
2.1 Conceptual design data preparation
2.2 Selected LSS conceptual documentation

Figure 5.3.1-1.- Program flow: Mission case study life support conceptual approach.

b. Due to the speculative and conjectural nature of lunar and martian mission planning, it is impossible to closely define the capabilities of an advanced EVAS based on precise mission and task descriptions. Therefore, the EVAS should be defined to impose as few limits on the EVA crewmembers as practical, to maximize the capability of the crew.

c. The Apollo lunar surface experience provides a data base from which to define generic lunar EVAS operational requirements.

d. Commonality of hardware and technology between Space Station Freedom, lunar, and Mars systems will be considered during this study.

Approach. In FY 1988, prior to definition by OEXP of the advanced mission scenarios, two independent studies were initiated to examine top-level EVAS requirements for typical lunar and Mars missions. Results from these efforts have been and will be used as starting points for further detailed examination to determine EVAS requirements for the OEXP mission case studies. A summary of the approach is listed below.

a. Review and critique lunar/Mars mission studies.

b. Derive generic mission requirements for the respective EVAS, based on the review of mission studies.

c. Identify unique EVAS drivers, based on mission, environment, or operational factors, for each case study.

d. Define strawman EVAS requirements to meet the generic mission requirements and unique drivers, for each case study.

e. Assess currently available EVAS capability and identify areas in which technology must be developed to meet derived EVAS requirements.

Findings. Since the lunar case studies were only recently completed and studies involving the martian system have just begun, current findings are limited to the lunar EVAS.

The lunar EVAS can be divided into five categories: pressure envelope, life support system (LSS), support vehicles, EVA support equipment, and airlock.

Pressure Envelope. The pressure envelope required for the lunar EVAS will be an anthropomorphic space suit assembly which will allow the suited crewmember, with life support system, to stand, walk, and kneel on the lunar surface (.165-g). Included among the pressure suit requirements will be the capability to regain footing after a fall. Maximal dexterity practicable should be provided in the suit gloves through use of special glove designs and minimal suit pressure.

The suit should provide protection from the lunar environment to the crewmember. Special attention must be paid to the problem of abrasion and contamination caused by the lunar dust. The suit must possess a long service life and be highly abrasion resistant, easy to clean, and rugged.

The pressure suit must be sized at the lunar base to fit from a 50th-percentile female to a 95th-percentile male. It must also be maintained at the lunar base, with no Earth depot maintenance required during its service life.

Life Support System. The LSS must provide atmospheric control for a nominal 8-hour EVA. It must be conveniently serviceable between uses, rugged, and reliable, and must possess an extended service life. It must be maintained at the lunar base, with no Earth depot service required during its operating life.

The life support system must be compact and of low mass, sufficient to allow ease of crewmember mobility without requiring excessive effort to maintain or reestablish center of gravity or balance and to prevent crewmember fatigue.

Support Vehicles. Ground transport vehicles are required for crew and equipment transfer to sites remote from the lunar habitat and for exploratory traverses. Support vehicles are also required for material handling at the habitat and at remote worksites.

EVA Support Equipment. EVA support equipment comprises the generic and specialized tools used by the EVA crew, as well as ancillary equipment (such as lights and television cameras) and the solar flare shelters used at remote sites.

Tools used by the EVA crew should, as much as possible, be selected from a generic tool kit containing a wide range of tools designed specifically for lunar EVA use. Specialized tools for individual tasks should be avoided but may be used when a generic tool capable of performing the task does not exist. A set of ancillary equipment should be provided to complement the generic tool kit. Included in the ancillary equipment would be floodlights; motion-picture, still, and video cameras; broad-use sensors and test equipment, etc.

Solar flare shelters must be provided at worksites far enough from the main lunar habitat that assured return to the habitat within the assumed warning time cannot be guaranteed. These must be permanent shelters, since expedient shelters will not provide a sufficient degree of comfort and habitability for the worst-case 96-hour shelter requirement.

Airlock. The airlock provides the interface between the lunar habitat interior and the lunar surface environment. It should provide passage for the EVA crewmembers to and from the lunar surface with minimal disturbance to the interior pressure of the habitat. The airlock also must provide isolation for the habitat interior from lunar surface dust.

Issues/Open Items. Each of the EVAS areas, as delineated above, has issues and/or items requiring further study. A short list in each category is presented below and summarized in table 5.3.2-I.

Pressure Envelope.

a. The need/utility for a hand-in capability versus technological, design, and operational difficulties of providing such a capability

b. Mobility/dexterity gain realized by lower suit pressures versus difficulties in habitat design due to lower habitat pressure forced by zero prebreathe requirement

c. Provision of removable coverall for lunar extravehicular mobility unit (EMU) for control of lunar dust contamination versus design and operational difficulties induced by such a garment, and compared to results obtainable with cleaning apparatus at the airlock

d. Actual requirements for solid waste handling and/or vomitus handling in-suit versus design and operational difficulties induced by such capability, and compared to alternate methods of dealing with each problem

Life Support System.

a. Use of regenerable versus open-loop consumables and impacts on LSS mass/volume and lunar base logistics

b. Use of 8-hour versus partial-day recharge/replacement of consumables in LSS and impacts on LSS mass/volume, design, operational safety, and lunar base logistics

c. Requirement/utility of "buddy system" connection between two LSS's for emergency use versus design difficulties and operational safety concerns induced by such a connection

TABLE 5.3.2-I.- LUNAR EVAS ISSUES/OPEN ITEMS—FY 1988 STUDY

Pressure envelope
- Hand-in capability
- Suit and habitat pressure
- Dust protection
- Waste handling

Life support system
- Regenerable vs open loop
- Partial day recharge
- Buddy system umbilical

Airlock
- Atmosphere recovery
- Volume
- Dust isolation measures
- EVA support function provided

Support vehicles
- Capabilities (crane, forklift, etc.)
- Performance (speed, range, etc.)
- Mission phasing of capabilities/ performance
- Pressurized cabin
- Ambulance provision
- Garage provision

EVA support equipment
- Solar flare shelter design
- Portable power supply availability

Support Vehicles.

a. Capabilities required of support vehicles. Included is the possible use of telerobotic vehicles for scouting and for nominal equipment transfer/materials handling

b. Required support vehicle performance (range, top speed, handling, cargo capacity, etc.)

c. Phasing of capabilities and performance of vehicles

d. Requirement for manned vehicle cabin pressurization versus design, cost, and operational difficulties

e. Need for ambulance vehicle/module (including pressurizable trauma treatment capability, mobile hyperbaric chamber, etc.) if any, versus cost, logistics, and operational difficulties

f. Need for storage facility ("garage") for support vehicles to provide protection against thermal shock at sunset and sunrise, micrometeoroid damage, and solar flare radiation

EVA Support Equipment.

a. Minimum requirements for solar flare shelter design, including the need for a permanent pressurizable shelter at each remote worksite versus expedient-type shelters requiring the crew to spend up to 5 days in pressure suits

b. Requirements for portable power supplies, including performance requirements and basic design approach (i.e., solar, fuel cell, batteries, nuclear)

Airlock.

a. Amount of atmosphere to recover/recycle versus cost and design difficulties and compared to pressurant logistics

b. Volume of airlock versus number of EVA crew to accommodate in airlock at once

c. Efficacy and characteristics of dust isolation measures

d. EVA support functions located in airlock versus dedicating airlock to environment interface only

FY 1989 Activities. The review of lunar mission studies and derivation of generic mission requirements will be completed in FY 1988. In FY 1989, unique lunar EVAS drivers will be identified and strawman EVAS performance and design requirements will be defined. The major vehicle for accomplishing this work will be trade studies to resolve the 19 issues and open items defined above. Current capabilities will be assessed to identify areas where technology must be developed to meet identified requirements.

In FY 1989, as results of the two martian system mission studies become available, the above process will be repeated for the Phobos and Mars surface mission scenarios. Completion is expected in late FY 1989.

SECTION 6

FY 1988 Exploration Study Observations

The purpose of this section is to provide summary information about the objectives of the FY 1988 case study process and then to identify and sort the findings of this year's activity into several categories which may be useful in planning for future exploration studies. These categories include an assessment of the sensitivity of the FY 1988 case studies to the parameters used in shaping their objectives and implementation ground rules; a comparative assessment of case study implementation concepts; and a collection of key functional area observations which, in general, have relevance across multiple case studies.

An important exploration studies "axiom" has surfaced during this first study year:

> *As insights into the technical systems and human "systems" performance needs for space exploration mature over the next several years, it is imperative that a broad array of implementation options be preserved or enabled at the strategic (multiprogram planning), tactical (mission planning) and execution (systems operations) levels.*

The synthesis of the FY 1988 case study prerequisite requirements with NASA's major development programs substantiated the validity of this axiom; ensuing key observations are summarized in this section. It was readily apparent that if any human exploration initiative is to be implemented, NASA must enable and build foundations in transfer vehicle and surface systems technologies, space and life science research, and Earth-to-orbit (ETO) and low-Earth orbit (LEO) systems and capabilities. As various parametric sensitivities and limitations are uncovered in these areas, and as case study objectives and ground rules are adjusted to better accommodate these sensitivities, having a "shopping list" of feasible implementation options at each level (each with its own unique performance and operations characteristics) will enable convergence in the early 1990's on an exploration initiative that will have valuable and significant yield early in the first decade of the next century.

6.1 STUDY OBJECTIVES OVERVIEW

The primary objective for FY 1988 was to study a set of potential exploration pathways (case studies) with a consistent methodology and with a uniform level of detail to compare and contrast diverse exploration strategies: expeditions, scientific outposts, and evolutionary approaches. The purpose of this effort was to determine the major factors and sensitivities that influence results in terms of the case study scale, complexity, benefits, and feasibility. For example, the scope and potential of various mission designs can realize a significant advantage through the use of advanced technologies and extraterrestrial resources; the assessments performed in this year's study cycle sought to determine the degree to which this is true. Another study factor included cost indicators (such as LEO mass) and the complexity associated with a single ETO launch compared to the benefit of a node in LEO for assembly activities.

To develop a strong knowledge base of exploration strategy sensitivities, case studies were selected to encompass a broad spectrum of objectives, requirements, and capabilities. The mission strategies range from a one-mission, expeditionary approach to a long-term evolutionary approach. For some cases, the systems and technology needs include those that are likely to be available in the near term; others assume the use of highly sophisticated new developments. ETO delivery systems requirements for the amount of mass that must be lifted to LEO range from 250 metric tons in the peak year for the Lunar Observatory, to seven times as much mass, 1,770 metric tons in the peak year, for the Mars Expeditions, which is prohibitive. The impracticality of this requirement is addressed later in this section.

The selected group of studies lands human explorers on the surface of another world anywhere from 2003 to 2014, with planetary surface stay times from as little as 14 days to almost 2 years. Gravitational conditions generate a unique range of requirements: on Phobos, the gravity is nearly zero; on the Moon, it is one-sixth that of Earth; on Mars, the gravity is one-third that of Earth; and during transit, it can be zero. The studies also cover a wide variety of trajectory profiles, number, and frequency of ETO and space transfer vehicle flights and mission durations.

6.2 CASE STUDY RESULTS COMPARISON

As a result of this process of developing a broad spectrum of strategies and approaches, a fairly extensive base of information has been developed that has enabled some new insights to be gained. One key finding from this year's studies is that the strategies and approaches employed in some case studies were good choices, and in other case studies were bad choices. For example, for the Mars expeditions, the choice of mission profile, transfer vehicle and surface habitat mass and volume, and propulsion system, etc., drove the mass in LEO requirement to values that are prohibitive. While the scenario employed for the Mars expeditions turned out to be a bad choice, having that result to add to the base of information is important. The "lessons learned" over the past year will be applied, in a continuing process of study, to the redirection and definition of future work.

6.2.1 Shaping Parameter Sensitivities

At the outset of this year's case studies, the key parameters that shape results were understood to be (1) ETO transportation; (2) LEO assembly and operations; (3) technology, including concepts of utilizing extraterrestrial resources; and (4) other factors that are unique to each case. A summary of these parameters as they were structured for study in FY 1988 is shown in table 6.2.1-I. What was learned about these shaping parameters from the case studies, special assessments, and trade studies is sorted into the same functional areas and summarized in the following sections

ETO Transportation. A dependable, high performance ETO transportation capability is of fundamental importance to the success of any exploration initiative. New capabilities will be required to enable timely delivery of massive space transfer vehicles, propellant, mission payload components, and support hardware to LEO for assembly and checkout. For instance, the Human Expedition to Phobos will require large amounts of mass to be lifted to orbit on each ETO flight, to minimize the number of elements that are assembled in LEO. On the other hand, the Mars expeditions (Case Study 2) could be accomplished with smaller ETO vehicles, since a robust onorbit assembly capability was assumed. However, issues related to ETO launch frequency and the availability of the LEO node then become important for Case Study 2.

Figure 6.2.1-1 illustrates the annual mass to LEO delivery requirements. This mass flow is of fundamental importance, since it directly affects the nature of the required ETO delivery system and Earth-orbital support facilities; furthermore, it is a rough indicator of total cost. The expeditionary approach is characterized by large peaks in mass, corresponding to the year chosen for launch. (As previously stated, the mass results for the Mars expeditions are excessive and the mission scenario employed in that case study will require reshaping in FY 1989.) In contrast, both the Lunar Observatory and the Lunar Outpost to Early Mars Evolution cases are characterized by steady rates of much lower magnitude, but over an extended period of time.

TABLE 6.2.1-I.- CASE STUDY SHAPING PARAMETERS

Parameters **Case Studies**	**Earth-to-Orbit Transportation**	**Low-Earth Orbit Assembly**	**Technology**	**Other**
Expeditions Strategy **Human Expedition to Phobos**	Large mass per launch	No assembly node	Use then-existing technology	Earliest mission
Human Expeditions to Mars	Large mass per year	Significant assembly at node each mission	Moderate increase in technology	Significant precursors
Science Outpost Strategy **Lunar Observatory**	Minimum mass per year	Minimum assembly at node each year	Moderate increase in technology	High science return
Evolutionary Strategy **Lunar Outpost to Early Mars Evolution**	Approximately constant mass per year	Moderate assembly at node each year	Advanced technology	Extraterrestrial resource usage

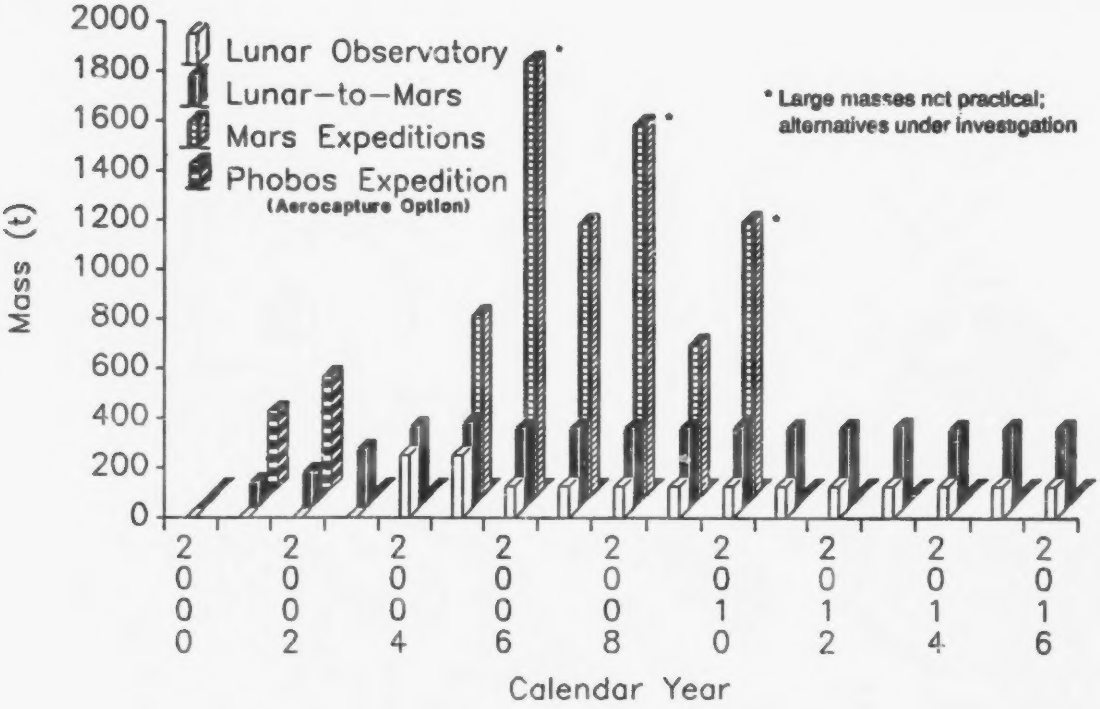

Figure 6.2.1-1.- Case studies mass summary—annual mass to LEO requirement

Because of the lack of maturity projected for onorbit assembly capabilities at the beginning of the next century, the earlier the mission (i.e., the Phobos case study), the stronger the case for minimizing such activities. This projection mandates the development of more capable ETO launch vehicles, in order to lift larger, but fewer, components to orbit. Heavy-lift launch vehicles (HLLV's) currently under study assume a mass-to-LEO capability on the order of 91 metric tons. With this performance, the baseline Phobos case requires as many as 30 separate launches, which is not practical. Innovative approaches must be sought to drastically reduce this requirement. One concept, to use the space transfer vehicle's integral propulsion system as an upper stage in the ETO stack to assist in the lift capability, warrants closer scrutiny. In addition to factors of cost and availability of ETO vehicles, the impacts on ground logistics for multiple flights per year are extensive. *A major decision that this nation faces is whether to invest in developing a heavy-lift vehicle that is at least twice as capable as those currently under study, to invest instead in a smaller lift capacity and develop an extensive capability for in-space assembly of large structures, or to effect a compromise between heavy-lift capacity and level of assembly in LEO.* The issue of ETO capability cannot be considered separately from the next functional area, LEO assembly and operations.

LEO Assembly and Operations. The choice of investment and mission strategies that affect LEO activities is a function of many interrelated variables. As presented in the previous section, it is recognized that there is a trade between onorbit assembly and ETO capability. (See figure 6.2.1-2.) In general, it is expected that the development costs of ETO transportation increase as the requirement for lift capability increases. On the other hand, as ETO lift capability increases, the expected trend is for the number of required onorbit operations (and assumed costs) to decrease, implying cost-optimum parameters exist that need to be identified.

Many of these issues were addressed in a cursory manner by trade studies and assessments in FY 1988, and will continue to be analyzed in more depth in FY 1989. However, some observations can be made, based on the preliminary results. Given our current experience base in space assembly operations, it is extremely difficult to project the level of operations required for these case studies. Obviously, lessons learned while we assemble Space Station Freedom will be extremely important; however, that experience is yet to come. Many of the assembly issues encountered during Freedom's ongoing design and development activity are common to the case studies as well, and the results may be applicable. Space Station Freedom's assembly planning has already encountered constraints imposed by the ETO systems (Shuttle) and by the crew (EVA time).

An important lesson learned from the Space Station

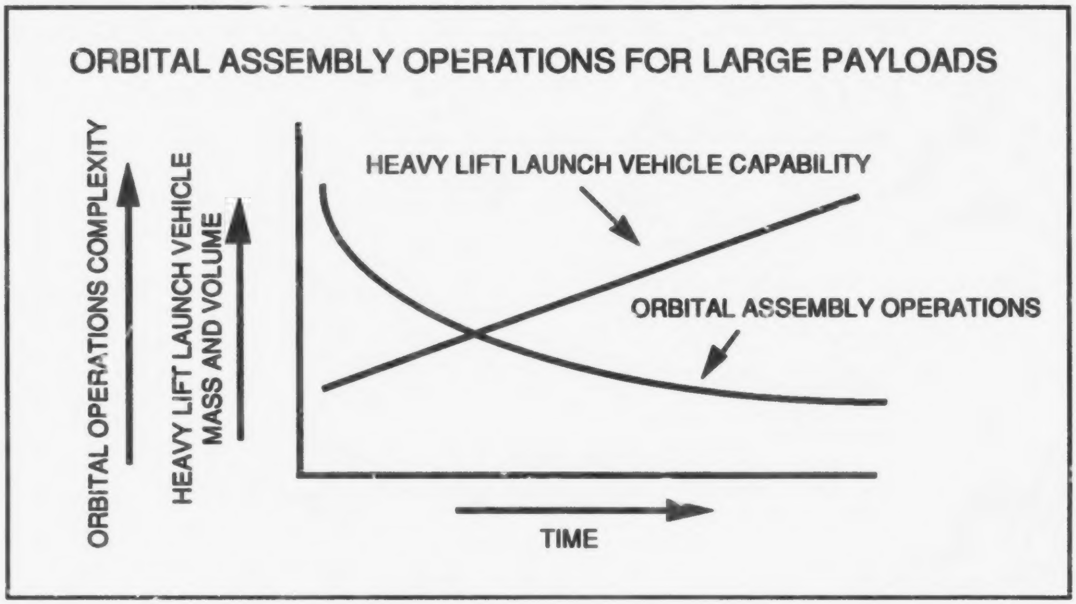

Figure 6.2.1-2.- Orbital assembly operations for large payloads.

Freedom experience is that the ETO support functions must commit to transportation performance stability in terms of agreed-to ETO performance. Degradation in delivered vehicle performance from concept definition to flight status could have potentially significant impacts on ongoing activities, since the development of the space transfer vehicles would begin before ETO vehicle flight readiness.

Equally important will be the level of automation and robotics technology available at the time to assist with, or perhaps perform all of, the assembly operations. Current approaches defined in the case studies require substantial human resources in LEO and as a result of the impact on ETO transportation, these approaches affect case study objectives negatively. Our special assessments and examination of prerequisite technology programs indicate that complex, detailed assembly operations will not be ready for robotic application in time to support our earlier (i.e., 2000) need dates. Thus, our current ground-based knowledge argues for minimizing and/or simplifying space assembly operations for the early missions; e.g., the Human Expedition to Phobos. However, some of the other cases, such as the Lunar Outpost to Early Mars Evolution study, envision advanced, reusable vehicles and the use of lunar liquid oxygen (LLOX), which implies a high degree of readiness for in-space operations technologies only a few years later. Since this particular case evolves over a longer period of time, it can and must be integrated with development programs. This is consistent with the forward-thinking philosophy employed in the development of the evolutionary case.

Of the four case studies under consideration this year, only the first, Human Expedition to Phobos, was baselined for study without the use of a LEO node. However, the baselined propulsive capture system resulted in such large masses in LEO that it is unlikely that it should be flown in that configuration. Because of these large masses, the option to launch a fully integrated (all-up) space transfer vehicle on a single ETO launch was considered impractical given current and projected ETO capabilities. Therefore, the very large masses required in LEO result in up to 30 ETO launches, and the integration activity of so many pieces in LEO exceeds the capability of a mate-and-dock approach.

To reduce the LEO mass to more manageable levels, an aerobrake capture system option was analyzed for the Phobos mission. This option resulted in about one-half the LEO mass of the baseline case and unless otherwise noted, the results of the aerobrake option for the Phobos Expedition are reported here. The technique of assembling the large aerobrake in LEO is not well understood, however, and it is unknown at this time whether a LEO node would be required to assemble the aerobrake.

Alternative crew module aerobrake designs that could be smaller, such as non-reusable ablators, could alleviate this problem, as could the use of nuclear thermal rockets (NTR's). Reducing the number of crew members for the Phobos mission further decreases the mass in LEO requirement and the subsequent ETO launches and space assembly operations; however, this option was not evaluated in detail.

Innovative approaches, such as advanced space propulsion (e.g., NTR's), may be required to undertake a major exploration program without requiring a LEO node. Further study is necessary to understand, in any detail, whether this mission can best be accommodated with or without LEO support infrastructure.

The Mars Expeditions and Lunar Outpost to Early Mars Evolution Case Studies have been structured to permit a significant and recurring amount of activity in LEO, and, therefore, have resulted in very challenging node support and operations scenarios. To define the operations for in-space assembly and vehicle processing for these cases, the natural tendency has been to attempt to understand our current experience in ground processing and extrapolate it to orbital operations. However, current ground processing flows for space vehicles represent a resource and time requirement that becomes unrealistic when imposed upon onorbit operations. New ways to process space vehicles that are assembled/mated onorbit will need to be developed to reduce the LEO operations work load and make these cases viable.

The resolution to this challenge most likely will be a combination of revising the vehicle design, eliminating processing functions, maximizing vehicle ground processing of resource-intensive tasks, incorporating automation and robotics and other strategies to enhance productivity and capability, and reducing the number of onorbit operations, as well as reducing the initial scale (mass) of the mission to lower the onset of requirements. This strategy will require both a "bottom-up" and a "top-down" approach. The bottom-up approach (currently under way and reported on in section 4.4, **Low Earth Orbit Assembly Strategies**) is one that begins with the current vehicle ground processing flows. Each function currently being performed needs to be accounted for in some manner (e.g., not required, incorporated into design, integrated with other functions, etc.) as well as new ones identified for the reference vehicles. The top-down approach is to establish an assembly resource allocation to be levied as a requirement for the integration agents. Results from both methods would then be used for convergence of requirements.

Impacts of Using Advanced Technology and Extraterrestrial Resources. The use of one or more key advanced technologies can cause a significant reduction in initial mass to LEO (IMLEO) requirements. As stated earlier, the use of aerocapture for orbit insertion at Mars and Earth can reduce by one-half the IMLEO requirements of standard chemical propulsion for orbit insertion for split/sprint missions to Mars and Phobos. Advanced propulsion techniques such as NTR's can reduce IMLEO for the Phobos Expedition by one-half and the Mars Expedition by one-third. Using an electric, low-thrust cargo vehicle in the Lunar Outpost to Early Mars Evolution case can reduce IMLEO by one-third over a standard chemically propelled vehicle. Also, advanced NTR technology for the piloted vehicle offers the potential for simultaneously reducing trip time, mass to LEO, and the logistical complexity of the evolutionary case study. For the Lunar Observatory case, advanced energy storage could extend crew staytime through the lunar night, and could also eliminate an Earth launch, although power is only one of several issues which must be solved to allow extended crew staytime.

The use of extraterrestrial propellant is a potentially high-leverage technology, and it was incorporated into the Lunar Outpost to Early Mars Evolution Case Study. In this case, the use of propellant from the Moon and Phobos (coupled with the use of electric propulsion on the cargo vehicle) can potentially reduce the IMLEO by more than one-third compared to the use of all Earth-based propellants. Of course, there is the question of facility investment costs, which are not understood at this time. A significant study of the details of startup and long-term costs of LLOX production facility is required to properly evaluate all the options.

The special assessment study on power showed a substantial mass advantage for nuclear power technology when compared to photovoltaic/regenerative fuel cell technology. When this mass advantage is folded into the total integrated lunar surface systems, the end result is a total case study mass reduction at LEO of one-half to three-fourths, depending on mission configuration, for power levels greater than 500-100 kW.

These are only a few examples of how advanced technologies are enabling for some areas and enhancing for others. The objective is to stimulate the development of technologies for the case studies to build a solid technology base from which NASA can select to support a variety of missions. This technological maturity will allow additional manned missions to other planets.

The problem that must be addressed is that different technologies can compete for the same research and development (R&D) funds. The question to be answered is which is more advantageous to pursue, from the standpoint of development cost and risk. An example would be that aerospace plane materials technologies and systems designed for the Advanced Launch System program could reduce LEO access costs to the point that extraterrestrial propellants would not be competitive with Earth-delivered propellants in LEO. However, which technology development has a greater probability of becoming a reality in the planned need timeframe, which has less operations risk, and which has the lower development costs? Related issues must be understood better through future case studies and trades.

Even so, alternative technologies must be available for each case study or the mission may be in jeopardy. The alternative technology may mandate that more mass be required to support the mission; for example, propulsive orbit entry as opposed to aerocapture may be required. For all the critical technologies, it must be determined whether an alternative exists and what penalty the missions will incur if that alternative is used. By allowing for the development of the technologies with proper funding and scheduling, the use of alternatives can be minimized.

The use of advanced technologies in a program carries with it an element of risk if the technology is developed in series with its intended use in the program. To alleviate schedule impacts, development of enabling technologies must be initiated well in advance of their required use. Ideally, the technology should be at a technology readiness level that has been demonstrated in the laboratory or integrated into a hardware subsystem at the start of phase C/D. The technology readiness level at the start of phase C/D will depend on the perceived risk of the nonavailability of the technology to support the mission: the higher the risk, the higher the technology level required. One related issue is that major ground and/or flight demonstrations will be required for certain technologies, such as aerobraking, cryogenic fuel handling, closed-loop life support systems, fractional-gravity spacecraft prototypes, exploration vehicles (such as the Phobos excursion vehicle), and nuclear power systems. These demonstration projects require long lead times, and must be resolved prior to the initiation of system development. To preserve schedules, these projects must be initiated immediately, or we will always be 5 to 10 years away.

The Agency technology development programs were compared with the case study programs, and where these programs were incompatible, alternative solutions were chosen wherever possible and practical. This was not possible in all cases and outstanding incompatibilities remain in three areas: (1) propellant transfer, (2) nuclear electric propulsion (NEP), and (3) Mars-to-Earth aerobraking. Also, some technology areas are not addressed currently in Project Pathfinder and the Civil Space Technology Initiative (CSTI), and thus some addi-

LEGEND

▨ Pacing

	Case Study (CS) Technology Need Date			
CRITICAL TECHNOLOGY	CS #1	CS #2	CS #3	CS #4
• Cryogenic Fluid Management	▨ 95	97	▨ 95	96
• Automated Rendezvous & Docking	95	94	95	96
• Autonomous Rovers		94		06
• Mars/Earth Aerocapture	▨ 95	▨ 94/97	▨ 95	▨ 96
• Onorbit Assembly & Construction		97	▨ 95	96
• Surface EVA Suits		98	95	96/06
• Surface Power (Including SP-100)		94	▨ 95	▨ 96/06
• Advanced Chemical Propulsion	95	97	95	96
• Nuclear Electric Propulsion				▨ 96
• Insitu Propellant Production				▨ 96
• Advanced Life Support	SS*	SS*		▨ 99

(SS* = Must be successful)

Figure 6.2.1-3.- Critical Technology Assessment.

tional work will need to be accomplished. Figure 6.2.1-3 lists the critical technologies, along with their need date to support the case studies, and indicates current incompatibilities with the Office of Aeronautics and Space Technology's planned ability to support the case studies in the given timeframe. It is apparent that a majority of technologies are common across most of the case studies. This indicates that by developing a core set of technologies it is possible to preserve the decision options for a number of missions.

Other Factors. In addition to the parameters that pertain to all case studies, each also holds a particular emphasis that must be considered in the overall planning strategy. These emphases are strictly dependent on the case study scenario itself, and also on the basic strategy that isselected for exploration.

For example, if the major motivation that is to be stressed is to achieve the earliest and first human voyage to another planetary body, then the Human Expedition to Phobos, which could arrive as early as 2003, becomes an attractive option. However, if the strategic emphasis is on the facilitation of opportunities for lunar and geophysical and cosmic astrophysical research, the Lunar Observatory Case Study could be considered attractive. The Human Expeditions to Mars have substantial precursor and prerequisite research requirements, both in terms of robotic missions to characterize the planetary conditions, and life sciences research to determine the effects on human beings of long-term exposure to the environment of space. And the Lunar Outpost to Early Mars Evolution Case Study explores and exploits the use of many new technologies, including those that mine and refine resources on the Moon or Phobos.

Clearly, the set of parameters derived from the studies conducted in FY 1988 defines a complex and interconnected situation. Many elements must be identified, assessed, planned for, and developed in parallel to enable the success of any mission. At the very heart of case study development is ETO transportation and the need for an ambitious launch schedule and a stable of vehicles that includes the Space Shuttle, an HLLV, expendables, and other advanced systems. Also critical is the availability of Space Station Freedom or another platform in LEO for assembly, in addition to a heavy-duty LEO space tug. The technology development schedules that are assumed impact all case studies, and each choice favors a particular exploration strategy.

6.2.2 Key Case Study Implementation Concept Comparisons

Once the key parameters that drive the case studies have been identified, a quantitative comparative analysis across the full range of case studies becomes possible. From this analysis, the case studies can be reshaped to focus further study of exploration paths. Although each has unique requirements and capabilities, an across-the-board comparison can be made in such areas as arrival date, infrastructure requirements, initial mass, mass delivered to the final destination and the capabilities enabled by this mass, number of crew members, where they go, and what they can achieve when they arrive. In this way, a relative image of the complexity and capability of all the case studies begins to emerge. Table 6.2.2-I is a summary of case study characteristics; some of these characteristics were derived from the case study analysis, but most were input requirements.

Human Expedition to Phobos. Based on the timelines and assumptions of the current case studies, the Phobos mission is potentially the earliest to arrive of the four case studies. A number of factors unique to this mission contribute to this capability.

The small crew of four minimizes the crew-associated facilities and supplies that must be carried. The fact that the crew does not land on the surface of Mars also simplifies both the scenario and the requirements for the mission. Mars surface landing systems are needed only for equipment, greatly reducing the time required for exploration program development and for the supporting technology and precursor programs. Assuming aerobraking at Mars, these simplifications allow the Phobos mission to be accomplished for about 30 to 40 percent of the mass-to-LEO requirement of a single Mars Expedition. Aerobraking technology and onorbit cryogenic fuel handling are the major technology developments required for this expedition scenario; the aspects of long-duration human flight is a significant unknown to be resolved as well.

The Phobos mission could be an excellent interim step to a manned Mars landing mission. The robotic exploration of Mars will provide improved knowledge of the martian environment and regolith. The Phobos mission will provide a unique opportunity to perform a systems checkout and verification of flight hardware and environment without the increased difficulty of a Mars landing. These considerations will allow a Mars-class mission to be accomplished four and a half years before the first Mars landing of the Mars expedition case.

However, the very large LEO masses required for this mission and the potential problems associated with LEO assembly and ETO transportation are areas of concern needing resolution in order to accomplish the top-level goal of an early mission to the Mars system. In addition, alternative solutions, such as the option to use aerobraking or NTR, while greatly alleviating the mass problem, create another potential problem by requiring additional

TABLE 6.2.3.-I.- SUMMARY OF CASE STUDY CHARACTERISTICS

SCENARIO / DESCRIPTOR	HUMAN EXPEDITION TO PHOBOS	HUMAN EXPEDITIONS TO MARS	LUNAR OBSERVATORY	LUNAR OUTPOST-TO-EARLY MARS EVOLUTION	
				LUNAR PORTION	MARS PORTION
o TRANSPORTATION					
- TRAJECTORY PROFILE	Cargo: min. energy Crew: sprint	Cargo: min. energy Crew: sprint	Translunar	Cargo: low thrust Crew: translunar	Cargo: low thrust Crew: near fuel min
- NUMBER OF FLIGHTS	1 Cargo, 1 Crew	3 Cargo, 3 Crew	2 Cargo, 2 Crew (set-up); 1 Crew flt per year thereafter	continuing LEO/LLO & LLO/LS shuttles	1 Cargo, 3 Crew
o CREW SIZE	4 (2 to phobos surface)	8 (4 to mars surface)	4	8	8 (8 to Mars surface)
o TOTAL CREW TRIP TIME	440 days	440, 440, 500 days	≤ 20 days	≤ 1 year	35 to 45 mos.
o SURFACE STAY TIME	30 days in Mars orbit 20 days at Phobos	30 days in Mars orbit 20 days on surface	< 14 days on surface (daylight only)	≤ 1 year	1-2 years
o EVA's (6 hours per EVA; two crew per EVA)	4 eva's at Phobos	4 EVA's at moons 10 EVA's at Mars 10-km unpress. rover traverses	12 EVA's 10-km unpress. rover traverses	EVA's as required 10-km unpress. rover traverse	EVA's as required 10-km unpress. rover traverse and 100-km press. rover traverse
o MASS TO LEO PEAK YEAR	Peak: 453 t @ 2002	Peak: 1770 t @ 2006	Peak: 250 t @ 2004 and 2006	Peak: 345 t @ 2005	
o PROPELLANT MASS (t)	Cargo veh: 234 Piloted veh: 318	Cargo veh: 1796 Piloted veh: 3363	Cargo veh: 87 Piloted veh: 96	Total: 1660 (all flights, lunar & Mars)	
o USER ALLOCATION (t)					
- ORBITAL	n/a	12.5, 12.5, 6	7	3.3	12
				(cumulative)	
- SURFACE	7 t Total: 2 teleop explorers on Mars	15, 15, 15	17.5 / Cargo flt. 6.5 / Crew flt.	112	Mars - 58 Phobos - 10
o PROPELLANT PRODUCTION	n/a	n/a	n/a	Four LLOX plants (40 t each)	Phobos prop. plant (86 t)
o YEAR - 1ST HUMANS TO SURFACE	2003	2007	2004	2004	2014

technology development for both aerocapture and NTR and, perhaps, development and deployment of LEO systems for assembly of the aerobrake. Additional study is required to find the optimum mix of solutions to these derivative problems in order to maintain the early mission date.

Human Expeditions to Mars. The Mars expeditions will deliver a crew of eight to Mars, with four landing on the surface. The Mars expeditions will require significant LEO infrastructure, including a five-fold increase in mass-to-LEO, and significant onorbit assembly operations at a LEO transportation node.

The Mars expeditions are more complicated than the Phobos expedition from several standpoints. Separate cargo and piloted vehicles must be built to land on (cargo and piloted) and ascend from (piloted) the Mars surface, significantly increasing the vehicular infrastructure complexity (and resultant IMLEO) for these missions. Mars EVA operations will require new pressure suits, portable life support systems, and surface transportation systems, which can safely and productively operate in the Mars one-third gravity, nonvacuum environment.

The LEO mass for these missions is unrealistically high and will require significant reshaping of the mission scenario in FY 1989. Since the integrated schedules for the Mars Expeditions have shown a launch date no earlier than 2007 due to complex systems development and the need to understand long-term health maintenance needs of humans in space, it certainly seems reasonable to revisit the ground rules of modest technology developments for this case study and push those technologies that can reduce the mass in LEO.

Lunar Observatory. The emphases of this case study are on science, constant operations learning experience, and minimizing infrastructure, accomplished by the use of teleoperated and man-tended facilities on the far side of the Moon. Significant human interaction will be required to assemble, deploy, operate, and service the array of instrumentation planned for this facility. In addition, once the facility is operational (in 2005), subsequent crews could be used for local geological exploration, as well as for scientific excursions in rovers for distances up to 10 km.

Peak year IMLEO is driven by case study infrastructure requirements. While the Lunar Observatory Case Study requires a substantial mass investment over its 10-year time horizon, due to the significantly less surface infrastructure required, its peak year mass in LEO is substantially less than the peaks in the Mars Expeditions Case Study.

The major driver for this scenario is the planetary surface activities requirements, including surface power systems, surface transportation systems, and EVA systems operations and maintenance. Long-term systems operations is an enabling capability to all of the Mars missions and the lunar cases that have permanent human presence. This implies the need for significant advanced development programs both on the ground and in space at Space Station Freedom. Certainly, the use of robotically assisted assembly and construction will be a promising new technology to investigate for the deployment of science facilities on the Moon.

Lunar Outpost to Early Mars Evolution. The major driver in this case is the development of advanced technologies, including systems that allow the exploitation of extraterrestrial resources for propellants. The use of in situ resources makes possible the accomplishment of significant lunar and Mars activities, at a much lower rate of annual mass expenditure. In addition, the evolutionary scenario has the added benefit of using its permanent Mars surface infrastructure to become even more efficient, with the further use of Mars resources, including propellants produced from materials on Phobos. The use of in situ resources results in a three-fold improvement in the ratio of useful hardware to propellant in LEO over the more conventional use of Earth-based propellants. Yet, key unknown areas identified in the study are the systems, technologies, and IMLEO requirements for mining, beneficiation, materials processing, storage, and transfer of in situ resources.

The conclusions that can be drawn from this year's studies are that a significant outpost is achievable with "small" masses (compared to those for expeditionary missions) and that this outpost is a good interim step to a permanent manned presence. However, many of the known systems and elements rely on the development of highly advanced technologies that are themselves still in immature stages of development and are subject to high risk of meeting the performance expectations used in the case study analyses. These technologies are critical to the success of this case study and are long-lead-time items; therefore, they should receive significant attention in the near term in order to achieve proof of concept and validate the case study performance estimates.

6.2.3 Functional Area Observations

This year's study activities have demonstrated the importance of the following key functional areas. All have the potential of being significant drivers in the design of human exploration missions, and for this reason further detailed analysis is required in the coming years to fully demonstrate their feasibility for the case studies.

Mission Trajectory Design. A fundamental aspect of human missions to Mars is the sensitivity of the Earth-to-Mars sprint-class trajectory to launch opportunity, as illustrated in figure 6.2.3-1. The IMLEO requirement can vary as much as 60 percent from opportunity to opportunity. This phenomenon can have a profound effect on spacecraft design resiliency to meet launch delays. The implications to program cost to design a common interplanetary transport capable of capturing the mission in several consecutive opportunities are enormous. Therefore, optimum launch opportunities must be protected, or study activities must be initiated to develop options for decoupling the mass performance from launch year for the Earth-Mars mission legs. Potential solutions are to use conjunction-class trajectories, which are less sensitive to celestial geometry, or opposition-class trajectories, which have intermediate performance demands.

Life Sciences Research. Throughout the FY 1988 studies, the issues of life sciences; i.e., advanced medical care, long-duration exposure to zero gravity, long-term exposure to the natural space environment (radiation), life support, and space human factors, have not been specifically addressed. Although these issues have been acknowledged, they have been assumed to be solvable in the timeframe under consideration; however, they can have significant impacts. In fact, the answers to the life sciences issues will be mission design drivers. Crew adaptability to zero gravity (or the need for artificial gravity) and space human factors will drive spacecraft design. The resiliency of the human body to varying gravity loads and radiation hazards throughout the mission will determine mission operations schedules and exploration sequencing. Knowledge in all of the life sciences areas is critical.

IMLEO and resupply logistics are a function of the degree of closure of the life support system. With the exception of lunar observatory systems, high closure of loop life support systems may provide such a large benefit that they may be enabling. Further analysis of the IMLEO sensitivity to life support system closure is required.

Nuclear Power Applications. Nuclear power concepts for both NEP and planetary surface applications have been shown to offer the potential for significant reduction of IMLEO. However, the conceptual designs proposed this year need further definition and study to

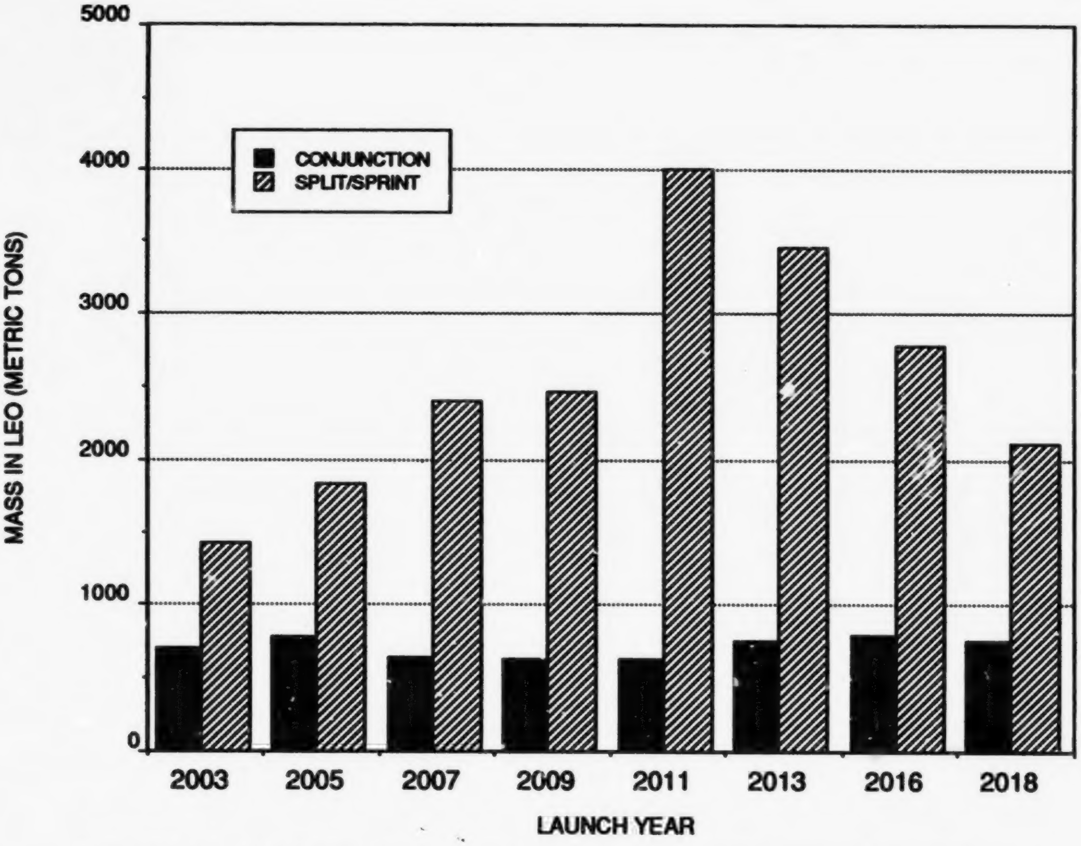

Figure 6.2.3-1.- IMLEO requirement sensitivity to launch year for human Mars missions.

demonstrate their feasibility. Case Study 4, for example, assumes a NEP cargo vehicle with an initial power level of 5 mW and specific mass levels of 5 to 10 kg/kWe for both lunar and Mars sorties. At this time, the feasibility of such multi-megawatt, lightweight nuclear power sources is an outstanding issue. These studies and ground-based system tests must be initiated to validate the electric cargo vehicle concept for the evolutionary case.

The FY 1988 study activities have assumed that nuclear reactors are viable power sources for planetary surface activities. The use of the SP-100 technology for these applications is explicit in Case Study 3, and implicit in Case Study 4. However, the SP-100 program in its current form as a space-based reactor does not enable surface power for either case study. Studies need to be performed to fully conceptualize these nuclear power systems for planetary surface applications. Also, the output would have to be extended to the multi-megawatt range for the NEP applications envisioned for Case Study 4.

The use of lunar He-3 to support nuclear fusion power on Earth could become the first truly extraterrestrial commercial venture with application and profit potential back on Earth. This is an area that demands a more detailed assessment. If the feasibility and practicality of mining lunar regolith to extract He-3 can be proven, this capability, in conjunction with a developing fusion technology, would have a far-reaching impact not only on human exploration missions, but also on global commerce.

Planetary Surface Systems Issues. The expedition to Phobos (or the Phobos portion of an expedition to Mars) presents the first opportunity for human exploration in the martian system without the associated complexity of landing upon Mars. However, Phobos possesses a unique set of environmental characteristics that make mission planning for human exploration of this moon extremely challenging. First, techniques for humans to remain on the surface in low gravity are undeveloped. The issue of surface mobility for human explorers relates to maintaining surface contact. A third area requiring more detailed study is the issue of dust particle contamination as a result of Phobos's surface conditions in combination with the low surface gravity. Insight into these issues, particularly the third area, may be forthcoming with results from the Soviet Phobos mission.

Case Studies 3 and 4 envision the eventual emplacement of permanently staffed habitats on the Moon and Mars. Much work is yet to be done to understand the surface operations associated with site selection, site preparation, and construction of these habitats and associated surface infrastructure. Much work is also required to understand lunar mining processes in order to support the assumed emplacement of the LLOX production facility in Case Study 4.

Issues of crew mobility and EVA-related activities in support of the above surface activities need further analyses. For example, the requirements levied on lunar and Mars crew members will severely test the current capabilities of spacesuit technology. In particular, lightweight spacesuits will be important for Mars and lunar surface applications due to the surface gravity environment.

Science Objectives Definition. The science objectives of the human exploration missions were derived through inputs from individual scientists, a few workshops, and the literature. Furthermore, this content has not undergone the scrutiny of a scientific oversight function. The scientific community at large needs to be involved in order to provide guidance to the definition of the case studies. Once objectives are more clearly defined and understood, they will be integrated into the case study mission scenarios.

6.3 IMPACTS ON PREREQUISITE PROGRAMS

During the previous year, the exploration study team has studied four exploration cases that were designed to bracket the many possible scenarios for human space exploration. The actual exploration initiative will most likely evolve to a combination of the studied cases, as well as others that may be formulated in subsequent years. Determination of the right mix will be the subject of debate among the users, the competing political and social priorities of the time, and the technicians who must provide the required capabilities. It has been learned during this year of study that NASA is not yet in a position to begin developing any of the actual exploration systems and elements, because a basic foundation needs to be laid in place. However, NASA has, through these case studies, learned much about the fundamental parameters that frame the structure of potential scenarios, although the feasibility of all that has been learned is yet to be demonstrated. It has identified the necessary prerequisite programs that must begin now in order to protect the opportunity to make a definitive exploration pathway decision in the early 1990's.

The needs of the case studies were compared with the current and planned activities of the NASA program offices in order to develop a basic understanding of the prerequisite foundations that will be necessary to protect the full range of strategies for exploration missions. Current approved programs were compatible with the case study needs in only a few functional areas. In almost every case, new programs are absolutely necessary to enable the proposed case study missions, and in a few functional areas, additional augmentation will be required. The case studies have also brought to light several functional areas, such as mining and construction, of which our understanding is not yet sufficiently mature to identify the support needs.

Advanced Transportation. In all cases, NASA's capability for launching cargo and personnel into LEO must be enhanced. Whether derived from current National Space Transportation System booster components or developed as a separate HLLV, this is absolutely enabling for any exploration initiative. The CSTI, managed out of the Office of Aeronautics and Space Technology (Code R) and in cooperation with the Office of Space Flight (Code M), has programs charged with the development of new booster technology, including propulsion systems and launch vehicles to enhance ETO transportation, and the investigation of aerocapture technology through the Aeroassist Flight Experiment. Both of these components of CSTI are critical to the human exploration initiatives.

Scientific Precursor Missions. Knowledge of planetary environments is required for the engineering design of the exploration systems and elements. The interaction of the exploration cases with approved and planned robotic missions managed by the Solar System Exploration Division (Code EL) must be thoroughly assessed. In particular the mission objectives of the Mars Observer, Lunar Observer, and Mars Rover Sample Return (MRSR) missions must be studied in order to optimize the support of both scientific and human exploration objectives. The Mars and Lunar Observer missions will reveal important precursor knowledge about the martian and lunar environments, respectively. MRSR, in addition to providing samples of the martian soil before the first human exploration, will also serve the important function of acting as a testbed for technology and operations demonstrations.

Humans-in-Space Research. Human health maintenance, long-term exposure to zero gravity, and life support systems technologies are critical to these exploration missions. Critical to this life science research is the humans-in-space (HiS) program of Project Pathfinder. The HiS program consists of three closely coordinated technological projects: (1) EVA/suit requirements definition and technology, (2) human performance (including space human factors, microgravity countermeasures, radiation effects and countermeasures), and (3) closed loop life support systems (including both physical/chemical and bioregenerative systems). The NASA Life Sciences Strategic Plan, as proposed by Code EB, must also be supported and augmented to accommodate the potential need for variable-gravity human research in LEO, radiation warning and protection systems, man-machine interfaces and human factors.

Advanced Technologies. The NASA Pathfinder advanced technology program managed by the Office of Aeronautics and Space Technology (Code R) is absolutely critical to the viability of these missions. Several key technologies addressed by Pathfinder critical to these exploration mission include (1) surface exploration, (2) in-space operations, and (3) space transfer, including high energy aerobraking. Also, advanced power systems (most likely nuclear), electric propulsion, and fluid transfer in space will still require augmentation to support the case studies. In addition, many of the planetary surface systems are not yet technically mature enough to identify all the needed technologies; therefore, additional requirements may be discovered in the next year of study.

Space Station Freedom. In all of these "bracketing" case studies, the need for a LEO assembly, servicing, and payload staging area was critical. A potential exception may be the Phobos expedition, although additional study is required to verify the ability to perform the mission in a reasonable and practical way without a LEO node. Also, a large amount of systems and technology inheritance, as well as an opportunity for basic technology development, was assumed to be provided by the Space Station Freedom program. This program, managed by the Office of Space Station (Code S), and its evolutionary planning and definition must stay on track. An important component of the Space Station Freedom program is the Evolutionary Working Group charged with studying potential enhancements to the phase I station to accommodate future mission requirements.

Communications and Data Tracking. Telecommunications, navigation, and information management (TNIM) capabilities will require upgrading to support the OEXP human exploration initiatives. There exist a number of options for the architecture of such support systems, and it is important that the Office of Space Operations (Code T) identify and analyze all candidate support options early, and that the implications of each option be understood for each exploration case study. The choice of options to pursue will depend on a number of factors, including the selected exploration scenario and its mission needs, the overall cost-effectiveness of the resulting architecture, NASA life-cycle costs, and the support needs of current unmanned NASA and international-cooperative missions at remote bodies.

APPENDIX A

LEXICON

LEXICON

This report uses a number of acronyms, abbreviations, and special terms. In order to facilitate the reader's comprehension of the text, a lexicon is presented here.

ACRONYMS AND ABBREVIATIONS

A&R - automation and robotics
A/C - aerocapture
AAS - American Astronautical Society
Ab or A_b - aerobrake
A_b/PROP - aerobraking/propulsion
ABDP - Aerobrake Design Program
ACS - Attitude control system
AEC - Atomic Energy Commission
AFE - Aerodynamic Flight Experiment
AFRPL - Air Force Rocket Propulsion Laboratory
AI - artificial intelligence
ALS - Advanced Launch System
ALSEP - Apollo Lunar Surface Experiment Program
Ant - antenna
ARC - Ames Research Center
Artificial-g - artificial gravity
ASAO - Advanced Space Analysis Office (located at the NASA/Lewis Research Center)
assy - assembly
ATCS - active thermal control subsystem
au - astronomical unit

BeO - beryllium oxide
Bio Med - biomedical

C - cargo
C - Centigrade or Celsius
C_3 - Square of the hyperbolic excess velocity in units $(km/s)^2$
C&T - communications and tracking
Car-1 - Cargo-1
Car-2 - Cargo-2
cc - cubic centimeter
CCD - charge coupled device
CDR - Critical Design Review
CELSS - controlled ecological life support system
CERV - Crew Emergency Return Vehicle
C_3H_8 - propane
CH_4 - methane
CHEBYTOP - Chebychev Trajectory Optimization Program
CIS - communications and information systems
CM - center of mass
CM - command module
cm - centimeter
CMG - control moment gyro
CMU - Carnegie Mellon University
C/O - checkout
CO - carbon monoxide

CO_2 - carbon dioxide
Code E - Office of Space Science and Applications
Code EB - Office of Space Sciences and Applications, Life Sciences Division
Code EL - Office of Space Sciences and Applications, Solar System Exploration Division
Code ES - Office of Space Sciences and Applications, Space Physics Division
Code M - Office of Space Flight
Code R - Office of Aeronautics and Space Technology
CODE RP - Office of Aeronautics and Space Technology, Propulsion, Power, and Energy Division
Code S - Office of Space Station
Code T - Office of Space Operations
Code Z - Office of Exploration
ComSat - communications satellite
ComSciSat - communications/science satellite
CP - center of pressure
CPM - Central Pressurized Module
CPM - Central Processing Module
CRYOTRAN - Computer code to predict fluid and thermal behavior of cryogenics in low gravity
CS-1 - Case Study 1
CS-2 - Case Study 2
CS-3 - Case Study 3
CS-4 - Case Study 4
CSI - control/structure interaction
CSM - (Apollo) Command Service Module
CSTI - Civil Space Technology Initiative
CTP - chemical transfer propulsion
CTV - cargo transfer vehicle
CV - cargo vehicle

D/He-3 - deuterium/helium-3
D/T - deuterium/tritium
DARPA - Defense Advanced Research Projects Agency
dB - decibel
dBi - decibels relative to antenna
DDT&E - design, development, test, and evaluation
DEV - Deimos excursion vehicle
DIAL - differential absorption light detection and ranging
DMS - data management system
DOD - Department of Defense
DOE - Department of Energy
DRL - Deimos robotic lander
DSM - deep space maneuver
DSN - Deep Space Network
DV or DV - delta velocity or mission velocity increment in units of kilometers/second (km/s)

E - estimated
EB - electron beam (welding)
ECCV - Earth crew capture vehicle
ECLSS - environmental control and life support system
ECV - electric cargo vehicle
ECV-RK - electric cargo vehicle refurbishment kit
EDCO - extended duration crew operations
EDO - Evolution Definition Office (located at Langley Research Center)
EDO - extended duration orbiter
EEI - Eagle Engineering Incorporated

EELS - Earth entry and landing system
EIRP - effective isotropic radiated power
ELV - expendable launch vehicle
EMG - Exploration Management Group
EMU - extravehicular mobility unit
EOC - Earth orbital capture
EOCS - Earth orbital capture system
EOS - Earth observation satellites
EP - electric propulsion
EPS - electrical power system
ERD - Exploration Requirements Document
ES - electrostatic
E-S Lib - Earth-Sun libration point
ESP - Exploration Study Plan
ET - exploration tools
ET - external tank
ET - extraterrestrial
ETM - Earth to Mars
ETO - Earth to orbit
ETP - extraterrestrial propellant
ETS - engine test stand
ETV - Earth transfer vehicle
ETX - Earth transfer expendables
EVA - extravehicular activity
EVAS - extravehicular activity system
EWG - Evolution Working Group
EXAC - Exploration Advisory Committee

F - thrust
FEL - first element launch
FOM - figure of merit
f/s - feet per second
fr/s - frames per second
FSI - flexible surface insulation
ft - feet
FTS - flight telerobotic servicer
FY - fiscal year

g - gram
g - gravity
Gb/s - gigabits per second
GC/NTR - gas core/nuclear thermal rocket
GCR - galactic cosmic radiation
GCR - gas core rocket
GDSS - General Dynamics/Space Systems (Division)
gee - acceleration of gravity at the surface of the Earth
GEO - geostationary Earth orbit (also geosynchronous Earth orbit)
GES - Geographic Entry System
GES - Ground Engineering System
GHz - gigahertz
GMEL - global manned exploration lander
GN&C - guidance, navigation, and control
GPHS - general purpose heat source
GSE - ground support equipment
GSFC - Goddard Space Flight Center
GSN - Geophysical Station Network

H_2 - hydrogen
Hab - habitation module
HAL - hyperbaric airlock
HC - hydrocarbon
HEO - high-Earth orbit
HiS - Humans-in-Space
HLLV - heavy lift launch vehicle
HMF - Health Maintenance Facility
HMO - high-Mars orbit
HP- human performance
HQ - Headquarters
hr - hour
HST - Hubble Space Telescope
Hz - Hertz

I/P - interplanetary
IA - integration agent
IDEAS - NASA Integrated Design and Evaluation of Advanced Systems computer software
IM - injection module
IMEO - initial mass in Earth orbit
IMLEO - initial mass to low-Earth orbit
IMM - interplanetary mission modules
IMS - ideal marginal savings
IOC - initial operational capability
I/P - crew habitat
I_{sp} - specific impulse (in units of seconds)
ISP - in situ propellant
ISPP - in situ propellant production
ISRU - in situ resources utilization
ISXP - in situ X production
ITV - interplanetary transportation vehicle
IVA - intravehicular activity

J - Joules
JEM - Japanese Experiment Module
JPL - Jet Propulsion Laboratory
JSC - Lyndon B. Johnson Space Center

K - temperature in Kelvins
Kb/s kilobits per second
kg - kilogram
kg/kW_j - propulsion system specific mass in kilograms per kilowatt of jet power
klbf - thousand-pound force
klbs - thousand pounds
km - kilometers
km^2/s^2 - kilometers^2/seconds^2
kN - kilonewton (1000 newtons)
KSC - John F. Kennedy Space Center
kW - kilowatts
kWe - kilowatt electric
kWj - kilowatt joules
kWth - kilowatts of thermal power

L - Earth launch
L/D - lift to drag ratio
L1/L2/L5 - Earth/lunar libration points
LaRC - Langley Research Center

LAV - lunar ascent vehicle
LB Model - Lunar Base Model
lbf - pounds-force
LBML - Lunar Biomedical Laboratory
LBS - lander body simulator
LCL - lunar cargo lander
LDCV - lunar descent cargo vehicle
LDPV - lunar descent piloted vehicle
LDR - large deployable reflector
LDRV - lunar descent rover vehicle
LDV - lunar descent vehicle
LDV-C - lunar descent vehicle - cargo
LDV-P - lunar descent vehicle - piloted
LEO - low-Earth orbit
LESC- Lockheed Engineering Sciences Company
LeRC - Lewis Research Center
LH$_2$ - liquid hydrogen
LHM - lunar habitation module
Lidar - light detection and ranging
LL - lunar lander
LLH$_2$ - lunar lquid hydrogen
LLO - low-lunar orbit
LLOX - lunar liquid oxygen
LMO - low-Mars orbit
LMO - lunar monitoring orbiters
LO - lunar observatory
LO - lunar observer
LOI - lunar orbit insertion
LOP - lunar O$_2$ plant
LOST - lunar O$_2$ systems tanks
LOV - lunar operations vehicle
LOX - liquid oxygen
LOX/LH$_2$ - liquid oxygen/liquid hydrogen
LPL - lunar piloted lander
LPO - lunar polar orbiter
LPV - lunar piloted vehicle
LPT - lunar propellant tanker
LRB - liquid rocket booster
LS - lunar surface
LSS - life support system
LTL - lunar tanker lander
LTM - local traverse mission
LTR - laser thermal rocket
LTV - lunar transfer vehicle

m - meters
MLL - manned lunar lander
MOTV - manned orbital transfer vehicle
Mab or Ma$_b$ - Mars aerobrake
MAR - mobility aids and restraints
MarsSciSat - Mars Science Satellite
MASE - Mission Analysis and System Engineering (JSC code Z support function)
MAV - Mars ascent vehicle
Mb/s - megabits per second
MCC - midcourse correction
MCL - Mars cargo lander
MCSV - Mars crew sortie vehicle

MCV - Mars cargo vehicle
MCVRCS - Mars cargo vehicle reaction control system
MDV - Mars descent vehicle
MELS - Mars entry and landing system
MEM - Mars excursion module
MEMU - Mars extravehicular mobility unit
MERI - Moon-Earth Radio Interferometer
MET - meteorological
MHR - Mars hyperbolic rendezvous
MHz - megahertz
Micro-g - microgravity
MIT - Massachusetts Institute of Technology
MLEO - mass to low-Earth orbit
MLI - multilayer insulation
MLL - Mars logistics lander
MLM - Mars landed module
MLMM - Mars landed mission module
MLOE - Mars landed operations equipment
MMH - monomethylhydrazine
MMII - Mariner Mark II
MMPG - Mars Mission Performance Generator
MMU - manned maneuvering unit
MMWAVE - millimeter wave
MO - Mars observer
MO - Mars orbit
MOC - Mars observer camera
MOC - Mars orbital capture
MOCS - Mars orbit capture system
MOI - Mars orbit insertion
MOO - Mars orbit operations
MOOS - Mars Orbital Operations System
MOR - Mars orbit refueling
MOSE - Mars orbit science equipment
MOV - Mars orbiter vehicle
MOX - Mars orbit expendables
MPD - magnetoplasmadynamic
MPD - magnetospheric particles detector
MPEM - Mars-Phobos excursion module
MPL - Mars payload
m/s - meters per second
MPV - Mars piloted vehicle
MRRAM - Mission Requirements and Resources Allocation Model
MRSR - Mars Rover/Sample Return
MSC - Mobile Servicing Center (an element of Space Station Freedom)
MSDB - Mission(s) and Supporting Elements Data base
MSFC - George C. Marshall Space Flight Center
MSS - Mars spaceship
MTM - Mars transfer module
MTR - Mars teleoperated rover
MTV - Mars transfer vehicle
MTX - Mars transfer expendables
MULIMP - Multi-Impulse Trajectory and Mass Optimization Program
MWAVE - microwave
mW_e - megawatts of electric power
mW_t - megawatts of thermal power

N - North
N_2O_4 - nitrogen tetroxide
N/A - not applicable
NAC - NASA Advisory Council
NAS - National Academy of Sciences
NaS - sodium sulfide
NASA - National Aeronautics and Space Administration
NEP - nuclear electric propulsion
Nerva - Nuclear Engine for Rocket Vehicle Application
NIH - National Institutes of Health
NLB - nuclear light bulb
nm - nanometer
nmi - nautical mile
NRC - National Research Council
NRX - Nuclear Reactor Experiment
NSTS - National Space Transportation System
NTO - nitrogen tetroxide
NTR - Nuclear Thermal Rocket

O_2 - oxygen
OAST - Office of Aeronautics and Space Technology (Code R)
ODSG - Orbital Debris Steering Group
OEXP - Office of Exploration (Code Z)
OMB - Office of Management and Budget
OMV - orbital maneuvering vehicle
Ops - operations
OSE - onorbit support equipment
OSF - Office of Space Flight (Code M)
OSO - Office of Space Operations (Code T)
OSS - Office of Space Station (Code S)
OSSA - Office of Space Science and Applications (Code E)
OSV - orbiting support vehicle
OTV - orbital transfer vehicle

PA - payload assist
Partial-g - partial gravity
PDR - Preliminary Design Review
PET - pulsed electrothermal thruster
PFC - primary (hydrogen/oxygen) fuel cell
Ph-Tele - Phobos teleoperator
PH/D - Phobos/Deimos
Phase II SS - Phase II Space Station Freedom
PhEV - Phobos excursion vehicle
PhSE - Phobos science equipment
PIC - power integrated circuit
PIT - pulsed inductive thruster
P/L - payload
PMC - permanent manned capability
PMS - precursor mission set
POTV - piloted orbital transfer vehicle
ppm - parts per million
PPP - Phobos pilot plant
PRD - Prerequisite Requirements Document
PRM - perigee raise maneuver
prox ops - proximity operations
PRP - (Pathfinder) Planetary Rover Program
PRR - Preliminary Requirements Review

psi - pounds per square inch (lb/in$_2$)
PSP - President's space policy
PSR - precision segmented reflector
PSS - planetary surface systems
PTD - Power Technology Division
PTF - propellant tank farm
PTL - Phobos tanker/lander
PTV - piloted transfer vehicle
PV - photovoltaic
PV - piloted vehicle
PVPA - photovoltaic power array
PV/RFC - photovoltaic/regenerative fuel cell

R&D - research and development
R&T - research and technology
RCS - reaction control system
rem - roentgen-equivalent man
Regen. GCR - regeneratively cooled, gas core rocket
RF - radio frequency
RFC - regenerative fuel cell
RM - radius of Mars
RMS - remote manipulator system
ROI - return on investment
RPM - retropropulsion module
RPPP - resource processing pilot plant
RSI - rigid surface insulation
RTG - radioisotope thermal generator
rvr - rover

S - South
s - second
S/C - spacecraft
SAA - special assessment agent
SAAP - sample acquisition, analysis, and preservation
SAIC - Science Applications International Corporation
SAR - synthetic aperture radar
SC/NTR - solid core/nuclear thermal rocket
SCR - solid core rocket
SD - solar dynamic
SDV - Shuttle-derived vehicle
SDWG - Scenario Development Working Group
SEP - solar electric propulsion
SETI - Search for Extraterrestrial Intelligence
SI - International System of Units
SiGe/GaP - silicon germanium/galium arsenide phosphide
SIM - simulation
SNPO - Space Nuclear Propulsion Office
SOA - state of the art
SOAR - Space Operations Analysis Resource software
SPE - solar-particle event
SPPD - Strategic Plans and Programs Division (Office of Space Station)
SR - sample return
SRD - Study Requirements Document
SRGCR - space radiator-cooled, open-cycle gas core rocket
SRM - solid rocket motor
SS - Space Station Freedom
SSED - Solar System Exploration Division

SSME - Space Shuttle Main Engine
STAR - Space Transportation and Resources software
STARBASE - Study Analyses Results Data Base
STBE - Space Transportation Booster Engine
STME - Space Transportation Main Engine
STR - solar thermal rocket
STS - Space Transportation System
STV - space transfer vehicle
SWG - Scenario Working Group

t - metric ton (tonne, 1000 kg)
T/W - thrust/weight
TACOM - Tank Automotive Command
TBD - to be determined
$^{\theta}$burn - engine burn time
TCS - thermal control system
TDRSS - Tracking and Data Relay Satellite System
TE - thermoelectric
TEA - torque-equilibrium attitude
TEI - trans-Earth injection
TEIS - trans-Earth injection system
TI - thermionic
TIC - Technical Implementation Concept
TLI - trans-lunar injection
TMI - trans-Mars injection
TMIS - trans-Mars injection stage
TMSR - Teleoperated Mars Surface Rover
TNIM - telecommunications, navigation, and information management
TPS - thermal protection system

Um - unmanned
USCM - unmanned sample collection missions
UV - ultraviolet

V - velocity
Variable-g - variable gravity
VCS - vapor-cooled shield
VCS - Vapor Cycle System
Vinf - V infinity or hyperbolic excess velocity
VGRF - Variable Gravity Research Facility
VLA - very large array
VLBI - very long baseline interferometry
VLFA - very low frequency array
VSB - Venus swingby

w/ - with
W - watts
WBS - work breakdown structure
Whr/kg - watt hours per kilogram
Wt - weight
XE-P - Experimental Flight Engine Prototype

yr - year

Zero-g - zero-gravity
ZPR - Code Z Program Review

DEFINITION OF TERMS

α - Propulsion system specific mass in kilograms per kilowatt of jet power (kg/kw_j).

A/B - Pre-development phase of program.

Aerobrake - Aerodynamic brake for use in planetary atmospheres.

Aerocapture - A technique of capturing a heliocentric spacecraft into a planetary orbit, using an aerobrake.

Astronomical unit - The distance from the Earth to the Sun; approximately 150 million kilometers.

Beneficiation - Improving the chemical properties of an ore so that metal can be recovered.

C/D - Development phase of program.

C_3 - Injection energy; square of the hyperbolic excess velocity in units of $(km/s)^2$.

Case Study Domain - That portion of the collective case study environs that is given to each integration agent as the physical/functional boundary for analysis. There have been three domains defined for each initiative in the exploration studies: planetary surface systems; orbital nodes; and transportation systems. Domains are functionally and physically interactive and must include appropriate inter-domain interface points in order to collectively constitute the end-to-end case study functional model.

Cislunar - Of or in the region of space between Earth and the Moon.

Controlled Ecological Life Support System - A spacecraft life support system that continually recycles solid, liquid, and gaseous materials essential for human life.

Cryogenic propellant - Propellant that must be stored at very low temperatures, e.g., liquid hydrogen and liquid oxygen.

Deep space maneuver - Propulsive maneuver performed along an interplanetary trajectory.

Deep Space Network - NASA Earth-based interplanetary communications system.

Earth Crew Capture Vehicle - Small vehicle for crew Earth orbit capture and/or Earth Entry and Landing System.

Earth Entry and Landing System - De-orbit propulsion plus aerobrake plus parachute plus terminal propulsion plus guidance and control.

Earth flyby injection maneuver - Interplanetary trajectory injection technique whereby the spacecraft makes a powered flyby gravity-assisted maneuver at Earth to reach critical injection energy.

Earth Orbit Capture System - Earth aerobrake plus retropropulsion plus guidance and control, if required.

Earth Transfer Expendables - Propellant and other consumables.

Earth Transfer Vehicle - Configuration of Mars spaceship for Mars to Earth transport of crew.

Earth-to-Orbit Vehicles - Launch vehicles such as expendable launch vehicles, Space Transportation System, and heavy lift launch vehicles.

Earth/Lunar libration point - (also Lagrangian point) Critical point in Earth-Moon space, where a body at rest would remain unless disturbed by an external force.

Electric Cargo Vehicle - Unmanned cargo vehicle propelled by Nuclear Electric Propulsion System.

Element - Within a domain, pertains to an integrated hardware unit, comprised of multiple systems and subsystems, which independently or in conjunction with other elements enables the execution of one or more case study objectives. Examples include a lunar roving vehicle, a crew habitation module, and a liquid oxygen production plant.

Exploration Requirements Document - Publication produced by the Office of Exploration that levies the overall exploration themes and objectives to initiate the FY 1988 studies activities.

Extraterrestrial propellant - Rocket propellant produced by the extraction of the appropriate constituents from an extraterrestrial body's environment .

Extravehicular activity - Any human activity outside protective shirt-sleeve environment and requiring a spacesuit.

Flight Telerobotic Servicer - Teleoperated robot for Space Station Freedom.

Galactic cosmic radiation - Cosmic rays from outside the solar system.

gee - Acceleration of gravity at the surface of Earth.

Geostationary Earth orbit (also geosynchronous Earth orbit) - Orbit in which satellite remains over same point on surface of Earth – about 35,800 km above equator – revolving at same angular speed as Earth.

Heavy lift launch vehicle - Earth-to-orbit vehicle with payload lift capability greater than 90 t to low-Earth orbit.

Helium-3 - The isotope of helium with mass number 3; constituting approximately 1.3 parts per million of naturally occurring helium on Earth. In sufficient quantities, potential fuel for nuclear fusion reactors.

Hooks and scars - Term used in association with Space Station Freedom. Refers to design techniques that "scar" Freedom to facilitate future planned evolution.

Hydrocarbon propellant - Methane (CH_4) or other.

Implementation (Requirements) - Those activities that react to a set of requirements with specific analyses, studies, and trades directed toward the provision of specific products, which may include concepts, element/systems architecture, recommended configurations, and operating strategies and techniques.

in situ - Latin expression meaning "in place;" used to refer to extraterrestrial locations. One common usage is *in situ* propellant production, which is synonymous with extraterrestrial propellant production.

in situ X production - e.g., X is: P=propellant, W=Water, R=resources, C=consumables, F=Food.

Integration agents - The Office of Exploration integration agents are Level III implementation agents. The integration agents are responsible for the definition of integrated sets of elements within specific domains. The domains are:

1. Planetary Surfaces
2. Space Transportation (excluding Earth-to-Orbit)
3. Space and Orbiting Nodes

The integration agents act as conceptual definition agents for elements of infrastructure that support the scenario development activities. The definition activity of an integration agent matures with time, beginning with basic functional descriptions, and evolving to concepts and point designs, but, in general, will not go to the level of detail expected in phase B development programs. At that point in time, the development responsibility is passed from Code Z to the responsible NASA Headquarters program office.

Interplanetary Mission Modules - Habitat/Laboratory/Logistics modules for crew in space.

L1 - Libration point; critical point in Earth-Moon space where a body at rest would remain unless disturbed by an external force.

Launch stack - The completely assembled interplanetary transport vehicle plus all propulsion stages prior to the departure injection maneuver.

Low-Earth orbit - A circular orbit about Earth with an altitude of approximately 300 to 500 km.

Low-Lunar orbit - A circular orbit about the Moon with an altitude of approximately 100 km.

Low-Mars orbit - A circular orbit about Mars with an altitude of approximately 250 to 500 km.

Lunar day/night - Approximately 14 Earth days each. The Moon completes one revolution about Earth in approximately 28 days.

Lunar Observer - Robotic scientific mission to study geochemistry and climatology of the Moon.

Lunar transfer vehicle - Vehicle for transportation between low-Earth orbit and the moon.

M/C_dA - Ballistic coefficient; Mass/Drag Coefficient of Area (Frontal).

Magnetoplasmadynamics - the generation of electric current by shooting a beam of ionized gas through a magnetic field.

"Magnum" Heavy Lift Launch Vehicle - Earth-to-orbit vehicle with 200 to 250 metric ton capability to low-Earth orbit.

Mars ascent vehicle - The vehicle that is launched from Mars surface to Mars orbit.

Mars cargo vehicle - Logistics vehicle sent to Mars for cargo staging.

Mars descent vehicle - The vehicle that de-orbits to land on Mars.

Mars entry & landing system - De-orbit propulsion plus aerobrake plus parachute plus terminal propulsion plus guidance and control.

Mars landed mission module(s) - Habitat/Laboratory/Logistics modules for the surface of Mars.

Mars landed operations equipment - Science, Transportation, Construction, Manufacturing equipment on the surface of Mars.

Mars Observer - Robotic scientific orbiter mission to Mars, planned for 1992 launch.

Mars orbit operating system - Propulsion for Mars orbit maneuvers.

Mars orbit science equipment - Instruments for studies from Mars orbit.

Mars orbital capture system - Mars aerobrake plus retropropulsion, if required, plus guidance and control.

Mars orbiting vehicle - Vehicle configuration in Mars orbit.

Mars spaceship - The spaceship that is assembled in low-Earth orbit.

Mars transfer module - Habitat/Laboratory/Logistics modules for crew in space.

Mars transfer vehicle - Configuration during flight to Mars.

Mission Analysis and System Engineering - The Mission Analysis and System Engineering (MASE) is a Level II implementation function of the Office of Exploration. The MASE group will decompose the scenario requirements into collections of top-level, functional requirements that must be accomplished by the integration agents (IA's). The integration agents will develop concepts that implement these requirements and furnish this information to the MASE group for integrated systems synthesis and total scenario option evaluation.

The MASE will also develop scenario dependent study issues for the special assessment agents (SAA's) and, as results are available from the special assessment agents, will assess total scenario impacts.

Monomethyl hydrazine - Bipropellant fuel.

National Space Policy and Exploration Guidelines -

- The policy specifies that in conjunction with other agencies: NASA will continue the lead role within the Federal Government for advancing space science, exploration, and appropriate applications through the conduct of activities for research, technology, development, and related operations.

- Space Science - NASA, with the collaboration of other appropriate agencies, will conduct a balanced program to support scientific research, exploration, and experimentation to expand understanding of: (1) astrophysical phenomena and the origin and evolution of the universe; (2) the Earth, its environment and its dynamic relationship with the Sun; (3) the origin and evolution of the solar system; (4) fundamental physical, chemical, and biological processes; (5) the effects of the space environment on human beings; and (6) the factors governing the origin and spread of life in the universe.

- Space Exploration - In order to investigate phenomena and objects both within and beyond the solar system, the policy states that NASA will conduct a balanced program of manned and unmanned exploration.
    - Human Exploration - To implement the long-range goal of expanding human presence and activity beyond Earth orbit into the solar system, the policy directs NASA to begin the systematic development of technologies necessary to enable and support a range of future manned missions. This technology program (Pathfinder) will be oriented toward a Presidential decision on a focused program of manned exploration of the solar system.

    - Unmanned Exploration - The policy further directs NASA to continue to pursue a program of unmanned exploration where such exploration can most efficiently and effectively satisfy national space objectives by, among other things, achieving scientific objectives where human presence is undesirable or unnecessary, exploring realms where the risks or costs of life support are unacceptable, and providing data vital to support future manned missions.

Nitrogen tetroxide - N_2O_4, bipropellant oxidizer.

Nuclear electric propulsion - Low-thrust electric propulsion, with electric power provided by nuclear reactor.

Nuclear Engine for Rocket Vehicle Application - (Nerva), nuclear thermal rocket program.

Nuclear light bulb - A type of open cycle gas-core nuclear thermal rocket; offers potential for perfect containment of both fuel and fission products through the use of a cooled transparent wall structure.

Nuclear Thermal Rocket - A space propulsion concept technique in which the heat from a nuclear fission reactor is used to raise the temperature of the propellant, which is then expanded through a nozzle to provide thrust. Two types of nuclear thermal rockets have been studied: gas core and solid core.

Office of Exploration Case Studies - Studies are specific mission scenarios that execute the exploration goals according to the objective content of the themes and strategies. Each case study may contain several optional implementation approaches. The case studies will be initiative-specific; each case study and its optional implementation approaches will address a single strategy. The four case studies analyzed in FY 1988 are:

1. Human Expedition to Phobos
2. Human Expeditions to Mars
3. Lunar Observatory
4. Lunar Outpost to Early Mars Evolution

Office of Exploration Strategies - The strategies present particular opportunities for meeting defined Office of Exploration themes. To organize and systematically examine a full range of options for human exploration and development of the Moon and Mars, three strategies were identified for study in FY 1988: 1) Expeditions, 2) Science Outpost, and 3) Evolutionary Expansion.

Office of Exploration Themes - The themes describe basic, upper-level objectives for space exploration, for example: national pride, advancement of scientific knowledge, etc. These themes provide a synthesis and a translation of the National Space Policy goals into a set of objectives that are compatible with the charter of the OEXP. These themes will be used to guide the generation of case study development requirements. The OEXP will produce and control the themes.

Orbital Nodes Integration Agent - The Office of Exploration Orbital Nodes integration agent is responsible for the definition and integration of all systems, elements and operational procedures for the low-Earth orbit transportation node, the lunar node, the Mars node and any other space or orbiting infrastructure required to support the exploration objectives.

Phobos Excursion Vehicle - Manned vehicle for transportation from Mars orbiting vehicle to Phobos.

Phobos Science Equipment - Instruments for studies of Phobos from a Phobos excursion vehicle.

Phobos Teleoperator - Remotely operated freeflyer to Phobos.

Photovoltaic Power Array - Power system operated by voltage generated as a result of exposure to visible or other radiation.

Planetary Surface Systems Integration Agent - The Office of Exploration Planetary Surface Systems integration agent is responsible for the definition and integration of all systems and elements and operational procedures for all infrastructure that resides on the surface of the Moon, Mars, or other planetary body.

Precursor Requirements - Science, technology, or operational data needed as critical path information to enable selection of specific habitation site location, location/objectives of specific user surface activities, systems design options, or specific operational approaches to human exploration. Precursor data are usually obtained via robotic, highly automated missions.

Prerequisite Requirements - A technical space system performance capability necessary for the execution of one or more exploration initiatives or scenarios. Prerequisite requirements are part of the exploration study and define case study-specific technology, space system, and operational support needs at a level of detail sufficient to enable the receiving program organization to proceed with its implementation strategy: either the development of new hardware elements, the modification of previously defined or existing hardware elements, or the use of existing hardware elements in support of the multiprogram initiative implementation effort.

Prerequisites Requirements Document - Publication produced by the Office of Exploration levying the required supporting precursor activities upon the other NASA Headquarters codes.

Propellant tank farm - Collection of propellant tanks for on-orbit fueling of interplanetary spacecraft.

Radioisotope thermoelectric generator - Self-contained power system in which a radioisotope is used to heat one junction in a circuit containing dissimilar metals, thus generating sustained electricity.

rem - A unit for measuring absorbed doses of radiation.

Remote Manipulator System - Space shuttle robot arm.

RL-10 - LH_2/LOX engine, manufactured by Pratt & Whitney.

Shuttle-C - Space Shuttle derivative proposed unmanned cargo vehicle.

Sol - A mean solar day for a given planet. Martian day; the modern term for the rotation period of Mars: 24h 37m 22.6s.

Solar electric propulsion - Ion drive; solar power; utilized in rocket systems; based on electric power, which can be derived from solar cells.

SP-100 - 100 kWe-class space power system.

Space Transportation System - All hardware systems, and support equipment, facilities, and manpower to deliver payloads to Earth orbit on board the Space Shuttle.

Space Vehicles Integration Agent - The Office of Exploration space vehicles integration agent is responsible for the definition, integration, and operations of all space vehicles beyond low-Earth orbit.

Special assessment agent - Directors of independent studies targeted towards the identification of high leverage technologies, systems, or operational techniques. Special assessment agents are truly independent and are not used as systems or subsystem definition agents for system designers.

Specific impulse - A performance parameter of a rocket engine expressed in seconds, equal to the thrust in pounds divided by the weight flow rate in pounds per second.

Stirling engine - An engine in which work is performed by the expansion of a gas at high temperature; heat for the expansion is supplied through the wall of the piston cylinder.

Strongback - Structural member that provides rigidity in bending and torsion.

Study Requirements Document - Publication produced by Mission Analysis and System Engineering levying the case study ground rules upon the integration and special assessment agents.

Subsystems - Within an element's particular system, pertains to those components which, when integrated together, form the functional hardware and software infrastructure of a system. Examples include power distribution, heat rejection, and data packeting subsystem.

Systems - Within an element, pertains to individual technical discipline areas which, when integrated together, form the functional hardware and software infrastructure of an element. Examples include power, thermal, and data management systems.

"______ System" - Pertains to one or more functionally independent or functionally interdependent elements within a scenario domain, the complement of which must operate in an integrated manner to achieve a major scenario objective. Examples include Earth-to-orbit transportation systems such as the National Space Transportation System or a class of Expendable Launch Vehicles, orbital transfer systems, such as the Orbital Maneuvering Vehicle and Orbital Transfer Vehicle, Low-Earth Orbit servicing systems such as the international space station, and planetary transportation systems such as the Lunar and Mars Rover. This term is usually used in conjunction with a word-set; e.g., National Space Transportation System, Advanced Launch System.

Technology Development - The development and demonstration of a hardware, software, or human capability with a performance beyond the current state-of-the-art. Examples of technology development in the Exploration Program are: space nuclear power generation (SP-100), automated rendezvous and docking, and medical health care for long duration missions.

Teleoperated Mars Surface Rover - Combined rover and sample return mission.

Teleoperator - A general-purpose, remotely controlled, cybernetic, dexterous person-machine system.

Telerobotic - Referring to automated systems operated remotely.

Trans-Earth injection - Mars orbital escape and trans-Earth.

Trans-Earth injection system - Propulsion and guidance system for trans-Earth Injection.

Trans-Mars injection - Earth orbital escape and trans-Mars.

Trans-Mars injection system - Propulsion and guidance system for trans-Mars Injection.

User - Any organization, group or individual who uses or plans to use the spacecraft, space elements, or space environs associated with the execution of an exploration initiative for scientific, technology development, or commercial application objectives.

APPENDIX B

PARTICIPANTS
AND
CONTRIBUTIONS

CONTRIBUTIONS

	SECTION TITLE	RESPONSIBLE AGENT *	AUTHOR (A) OR STUDY MANAGER (M) *
	VOLUME I: EXECUTIVE SUMMARY	MASE	SAIC/J. Soldner (A)
	VOLUME II: STUDY APPROACH AND RESULTS		
1.0	EXPLORATION STUDIES APPROACH	MASE	LESC/E. Smith (A)
2.0	EXPLORATION CASE STUDY RESULTS		
2.1	HUMAN EXPEDITION TO PHOBOS	MASE	JSC/S. Wilson (M)
2.1.1	Case Study Overview	MASE	JSC/S. Wilson (A)
2.1.2	Mission Definition and Manifest	MASE	JSC/S. Wilson (A)
2.1.3	Mission Architecture and Infrastructure	MASE	JSC/S. Wilson (A)
2.1.4	Transportation Systems Definition	Transportation IA	MSFC/F. Huffaker (M)
2.1.5	Orbital Node Systems Definition	Node IA	LaRC/B. Pritchard (M)
2.1.6	Planetary Surface Systems Definition	Surface Systems IA	JSC/J. Alred (M)
2.1.7	Case Study Synthesis	MASE	JSC/S. Wilson (A)
2.2	HUMAN EXPEDITIONS TO MARS	MASE	JSC/S. Wilson (M)
2.2.1	Case Study Overview	MASE	JSC/S. Wilson (A)
2.2.2	Mission Definition and Manifest	MASE	JSC/S. Wilson (A)
2.2.3	Mission Architecture and Infrastructure	MASE	JSC/S. Wilson (A)
2.2.4	Transportation Systems Definition	Transportation IA	MSFC/F. Huffaker (M)
2.2.5	Orbital Node Systems Definition	Node IA	LaRC/B. Pritchard (M)
2.2.6	Planetary Surface Systems Definition	Surface Systems IA	JSC/J. Alred (M)
2.2.7	Case Study Synthesis	MASE	JSC/S. Wilson (A)
2.3	LUNAR OBSERVATORY	MASE	JSC/L. Livingston (M)
2.3.1	Case Study Overview	MASE	JSC/L. Livingston (A)
2.3.2	Mission Definition and Manifest	MASE	JSC/L. Livingston (A)
2.3.3	Mission Architecture and Infrastructure	MASE	JSC/L. Livingston (A)

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* Organization symbols and other acronyms are defined in Appendix A. It is recognized that study managers (M) were often supported by one or more contractor organizations which provided authorship of data/reports within this document. Space does not permit a complete listing, but these technical services have been invaluable and very much appreciated.

	SECTION TITLE	RESPONSIBLE AGENT *	AUTHOR (A) OR STUDY MANAGER (M) *
2.3.4	Transportation Systems Definition	Transportation IA	MSFC/F. Huffaker (M)
2.3.5	Orbital Node Systems Definition	Node IA	LaRC/B. Pritchard (M)
2.3.6	Planetary Surface Systems Definition	Surface Systems IA	JSC/J. Alred (M)
2.3.7	Case Study Synthesis	MASE	JSC/L. Livingston (A)
2.4	LUNAR OUTPOST TO EARLY MARS EVOLUTION	MASE	JSC/E. Lineberry (M)
2.4.1	Case Study Overview	MASE	JSC/E. Lineberry (A)
2.4.2	Mission Definition and Manifest	MASE	JSC/E. Lineberry (A)
2.4.3	Mission Architecture and Infrastructure	MASE	JSC/E. Lineberry (A)
2.4.4	Transportation Systems Definition	Transportation IA	MSFC/F. Huffaker (M)
2.4.5	Orbital Node Systems Defiition	Node IA	LaRC/B. Pritchard (M)
2.4.6	Planetary Surface Systems Definition	Surface Systems IA	JSC/J. Alred (M)
2.4.7	Case Study Synthesis	MASE	JSC/E. Lineberry (A)
3.0	PREREQUISITE REQUIREMENTS IMPLEMENTATION PLANS	MASE	JSC/B. Roberts (M)
3.1	EXPLORATION IMPACTS TO ETO TRANSPORTATION	HQ Code M	Code M/B. Askins (M)
3.2	EXPLORATION IMPACTS TO UNMANNED SOLAR SYSTEM MISSION PROGRAMS	HQ Code EL	Code EL/W. Conway (M)
3.3	EXPLORATION IMPACTS TO SPACE LIFE SCIENCES	HQ Code EB	Code EB/M. Connors (M)
3.4	EXPLORATION IMPACTS TO ADVANCED TECHNOLOGY DEVELOPMENT	HQ Code R	Code R/W. Hudson (M)
3.5	EXPLORATION IMPACTS TO COMMUNICATIONS, TRACKING, AND SATELLITE SUPPORT SYSTEMS	HQ Code T	Code T/A. Milier (M)
3.6	EXPLORATION IMPACTS TO SPACE STATION EVOLUTION	HQ Code S	Code S/ E. Huckins (M)
3.7	CASE STUDY PROGRAM INTEGRATION SCHEDULES	MASE	JSC/A. Dula (M)

	SECTION TITLE	RESPONSIBLE AGENT *	AUTHOR (A) OR STUDY MANAGER (M) *
4.0	SPECIAL TRADE STUDIES AND REPORTS	MASE	JSC/J. Bell (M)
4.1	EARTH-MOON NODE LOCATION	MASE	SAIC/J. Soldner (A)
4.2	EXTRATERRESTRIAL PROPELLANT LEVERAGING		
4.2.1	JSC-Sponsored ETP Studies	MASE	JSC/K. Fairchild (M); LESC/D. Weaver (A)
4.2.2	In situ Propellant Leverage Analysis	Prop SAA	Analex Corp./ A. Willoughby (M); GDSS/B. Cordell (A)
4.3	FEASIBILITY OF AUTOMATING LLOX PRODUCTION	A&R SAA	ARC/M. Sims (M)
4.4	LOW EARTH ORBIT ASSEMBLY STRATEGY		
4.4.1	End-to-end Assembly	MASE	JSC/J. Bell (M) SAIC/J. Soldner (A, Strategic Plan); LESC/D. Weaver & Eagle/W. Stump (A, Launch Vehicle Configuration)
4.4.2	Onorbit Assembly vs. Ground Assembly Functions	Node IA	LaRC/B. Pritchard (M)
4.4.3	LEO Assembly Operations Support Systems and Techniques	Ops SAA	JSC/R. Trevino (M)
4.5	POWER AND PROPULSION PARAMETERS FOR NUCLEAR ELECTRIC VEHICLES	Prop SAA	LeRC/J. Riehl, L. Mason, J. Savey, H. Bloomfield (A's); Sverdrup Technology, Inc./ J. Gilland (A)
4.6	TELEOPERATED ROVERS IN SUPPORT OF HUMAN PLANETARY EXPLORATION	A&R SAA	ARC/M. Sims (M)
4.7	LUNAR OBSERVATORY EXTENDED CREW STAY POWER ANALYSIS	Surface Systems IA	LeRC/R. Cataldo, J. Bozek (A's)
4.8	PHOBOS EXPLORATION ASSESSMENT	Surface Systems IA	JSC/A. DuPont (A)
4.9	SPACE EXPLORATION COST UNDERSTANDING	Cost SAA	JSC/K. Cyr (A)
4.10	SCIENCE OPPORTUNITIES IN HUMAN EXPLORATION INITIATIVES	MASE	JSC/M. Duke (A)

	SECTION TITLE	RESPONSIBLE AGENT *	AUTHOR (A) OR STUDY MANAGER (M) *
4.11	POWER TECHNOLOGY WORKSHOP RESULTS	Power SAA	LeRC/M. Valgora (A)
4.12	LUNAR HELIUM-3 WORKSHOP RESULTS	Power SAA	LeRC/J. Hickman (A)
4.13	ADVANCED SPACE PROPULSION WORKSHOP RESULTS	Prop. SAA	LeRC/O. Spurlock (A)
4.14	ROBOTICS WORKSHOP RESULTS	A&R SAA	ARC/M. Sims (A)
4.15	MINIMUM CREW SIZE FOR PHOBOS & MARS MISSIONS	MASE	JSC/K. Fairchild (A)
5.0	IN-DEPTH SYSTEMS ASSESSMENTS	MASE	JSC/J. Bell (M)
5.1	POWER SYSTEMS	Power SAA	LeRC/M. Valgora (M)
5.1.1	SP-100 Nuclear Power System Conceptual Design for Lunar Base Applications		LeRC/L. Mason, H. Bloomfield (A's)
5.1.2	Solar Photovoltaic vs. Nuclear Power for Lunar Observatory		LeRC/J. Hickman, H. Bloomfield (A's)
5.2	PROPULSION SYSTEMS	Prop. SAA	LeRC/D. Schultz (A, M)
5.2.1	Evaluation of Advanced Propulsion/Power Concepts		LeRC/S. Borowski (A)
5.2.2	Impact of Solid-Core NTR Propulsion on Human Expeditions to Phobos/Mars		LeRC/S. Borowski (A)
5.2.3	Impact of Phased Implementation of Solid and Gas Core NTR Propulsion on the Evolutionary Lunar-to-Mars Outpost Study		LeRC/S. Borowski (A)
5.2.4	Issues of Mars Orbital Refuling		Analex Corp./ A. Willoughby (M); GDSS/B. Cordell (A)
5.2.5	Launch of Cryogenic Tanks		LeRC/R. Corban (A)
5.3	ADVANCED LIFE SUPPORT SYSTEMS	ALS SAA	JSC/A. Behrend (A, M) JSC/D. Price (A)
6.0	FY 1988 EXPLORATION STUDY OBSERVATIONS	MASE	JSC/B. Roberts SAIC/J. Soldner (A)
APPENDICES			
A.	LEXICON OF ACRONYMS AND TERMS		SAIC/T. Ramlose (A)
B.	PARTICIPANTS/CONTRIBUTIONS		JSC/D. Bland (A)
C.	LIST OF ANALYSIS TOOLS/FACILITIES		JSC/K. Fairchild (A)

APPENDIX C

FY 1988 OEXP SYSTEMS ENGINEERING AND DESIGN ANALYSIS TOOLS

Following are descriptions of computer tools used to generate and quantify the FY 1988 OEXP case studies. They include the level II systems engineering development tools used by MASE for case study development and level III detailed design and analysis tools used by Integration Agents (IA's) and Special Assessment Agents (SAA's).

Lunar Base Model (LB Model) - Large Scale Programs Institute (LSPI)

The Lunar Base Model is a series of Symphony (version 1.1) spreadsheets, controlled by sets of macros. The model follows a three-part structure: a missions data base, a set of infrastructure modules, and a transportation module. It is driven by the user's selection of an input missions manifest and various support technology options. The model performs calculations by a year-to-year or sequential method. Equation forms within the model vary according to the operation at hand. Methods of calculation include post-processing matrix manipulations, linear equations, sizing algorithms, conditional scheduling, and data accumulation. For a more detailed description of the current model, refer to the Lunar Base Model Reference Manual, version 2.2 release, 1988.

Multi-Impulse Trajectory and Mass Optimization Program (MULIMP) - Science Application International Corp. (SAIC)

The MULIMP program, which generates optimum ballistic interplanetary trajectories, was developed by SAIC under subcontract to JPL, and is thus in the public domain. MULIMP generates multitargeted trajectories as a series of two-body subarcs; it allows intermediate flyby, gravity-assisted, and deep-space impulse maneuvers. Originally written to run on the JPL UNIVAC 1100 series computer, the program has been run by SAIC for the past 5 years in a Digital Equipment Corp. (DEC) PDP minicomputer. SAIC has just completed a version compatible with PC MS-DOS systems.

Chebychev Trajectory Optimization Program (CHEBYTOP) - SAIC

The CHEBYTOP program generates optimum low-thrust interplanetary trajectories for either solar electric (SEP) or nuclear electric (NEP) propulsion systems. CHEBYTOP was originally developed by Boeing Aerospace in the late 1960's as a stand-alone subroutine. SAIC's low trust trajectory generator consists of a main program shell which incorporates version III of CHEBYTOP. The program includes analytical approximate solutions of planet spiral maneuvers. Originally written to run on the JPL UNIVAC 1100 series computer, the program has been run by SAIC for 5 years in a DEC PDP minicomputer environment. SAIC has just completed a version compatible with PC MS-DOS systems.

Mars Mission Performance Generator (MMPG) - SAIC

This analysis package consists of a series of spreadsheets written in EXCEL software for the purpose of generating piloted Mars mission mass performance estimates. The program accesses a trajectory data base of round-trip Mars missions or user-specified missions. The output from the program consists of a set of system mass and event delta-V summary tables. The program can currently evaluate 48 different mission types. This software tool was developed by SAIC under subcontract to Martin Marietta Aerospace, and is currently operating on Apple Macintosh II hardware.

Mission Requirements and Resources Allocation Model (MRRAM) - Martin Marietta (MM)

MRRAM is a Macintosh-based model to calculate primary mass allocations from power, volume, structure, orbital mechanics, margins, artificial-g, crew size, life support, etc.

Aerobrake design program (abdp) - MM

The abdp (not an official name) computes aerobrake thermal and dynamic loads profiles for a given aerobrake design to derive appropriate trajectories and resulting aerobrake mass and size.

NASA Integrated Design and Evaluation of Advanced Systems + Structural Dynamics Research Corporation (SDRC) Integrated Design Engineering and Analysis System (IDEAS^2) - Langley Research Center (LaRC)

This computer-aided engineering system performs engineering analyses (e.g. structural loads, thermal loads) on three-dimensional computer models.

The table below shows the sections of this document in which each of the tools described above were used.

TABLE C-I.- ANALYSIS TOOLS

VOLUME II SECTION		ANALYSIS TOOLS USED
2.0	EXPLORATION CASE STUDY RESULTS	
2.1	HUMAN EXPEDITION TO PHOBOS	
2.1.2	Mission Definition and Manifest	MULIMP, MMPG
2.1.4	Transportation Systems Definition	MMPG, MRRAM, abdp
2.1.5	Orbital Node Systems Definition	IDEAS^2
2.2	HUMAN EXPEDITIONS TO MARS	
2.2.2	Mission Definition and Manifest	MULIMP
2.2.4	Transportation Systems Definition	MMPG, MRRAM, abdp
2.2.5	Orbital Node Systems Definition	IDEAS^2
2.3	LUNAR OBSERVATORY	
2.3.4	Transportation Systems Definition	MMPG, MRRAM, abdp
2.3.5	Orbital Node Systems Definition	IDEAS^2
2.4	LUNAR OUTPOST TO EARLY MARS EVOLUTION	
2.4.2	Mission Definition and Manifest	LB Model, MULIMP MMPG, CHEBYTOP
2.4.3	Mission Architecture and Infrastructure	LB Model
2.4.5	Orbital Node Systems Definition	IDEAS^2
4.0	SPECIAL TRADE STUDIES AND REPORTS	
4.1	Earth-Moon Node Location	MULIMP, CHEBYTOP

A number of other analysis tools defined during a NASA-sponsored workshop held at LaRC in June 1988 will be useful to IA's and SAA's in future design and analysis of OEXP case studies. Detailed listings and descriptions of these tools are given in the workshop proceedings (L.B. Garrett, R.L. Wright, and D. Bodi, compilers: OEXP Analysis Tools Workshop. NASA CP-10013, August 1988.)

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16. Abstract

The Office of Exploration (OEXP) at NASA Headquarters has been tasked with defining and recommending alternatives for an early 1990's national decision on a focused program of human exploration of the solar system. The Mission Analysis and System Engineering (MASE) group, which is managed by the Exploration Studies Office at the Lyndon B. Johnson Space Center, is responsible for coordinating the technical studies necessary for accomplishing such a task. This technical report, produced by the MASE, describes the process that has been developed in a "case study" approach. The four case studies that were developed in FY 1988 include:

1. Human Expedition to Phobos
2. Human Expeditions to Mars
3. Lunar Observatory
4. Lunar Outpost to Early Mars Evolution

The final outcome of this effort is a set of programmatic and technical conclusions and recommendations for the following year's work.

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